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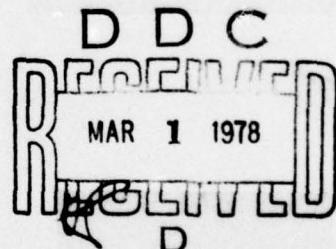
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INVESTIGATION OF A HIGHLY  
LOADED TWO-STAGE FAN-DRIVE TURBINE

VOLUME V. Phase III, Boundary Layer Control Optimization and Off-Design Evaluation

H. Welna, D. E. Dahlberg and W. H. Heiser  
Pratt & Whitney Aircraft  
Division of United Aircraft Corporation

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**FOREWORD**

- (U) This Interim Technical Status Report (Contractor's Reference No. PWA 3967) was prepared by Pratt & Whitney Aircraft, Division of United Aircraft Corporation, East Hartford, Connecticut, as the fifth Semiannual Report under United States Air Force Contract F33615-68-C-1208, Project No. 3066, Task No. 306606. This report was submitted by the Contractor on 30 June 1970, and covers the report period from 1 January 1970 to 30 June 1970.
- (U) The findings and conclusions of this report are not deemed as final by the Contractor. They are subject to verification or revision in the Final Report to be published upon the completion of this Contract.
- (U) The Air Force Program Monitor is Mr. Wayne Tall, APTC, Air Force Aero Propulsion Laboratory, Wright-Patterson Air Force Base, Ohio, 45433.
- (U) This report contains no Classified information extracted from other Classified documents.
- (U) Publication of this report does not constitute Air Force approval of the report's findings or conclusions. It is published only for the exchange and stimulation of ideas.

**Wayne Tall  
Project Engineer  
Air Force Aero Propulsion Laboratory**

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## **UNCLASSIFIED ABSTRACT**

(U) A comprehensive, four-phase, three-year program is in progress to investigate methods of improving the performance of fan-drive turbines. The goals of this program are to develop turbine design procedures and aerodynamic techniques for efficient, high work, low pressure turbines.

(U) The first phase effort of defining the preliminary turbine design has been completed and the results were reported (Reference 1). The second phase consisted of an experimental evaluation which included establishment of both two-dimensional loss levels and three-dimensional flow behavior for the baseline airfoils, and for airfoils utilizing various boundary layer control methods. The design of the baseline cascade packs was reported in Reference 2, and the annular cascade performance of the baseline airfoils was reported in Reference 3. The annular cascade results of the baseline airfoil boundary layer control methods were reported in Reference 4.

(U) The results of the plane-cascade, two-dimensional, baseline investigation, the decreased-solidity annular-cascade airfoil test results, and the Phase III boundary layer control optimization and off-design evaluation are presented in this report.

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## LIST OF SYMBOLS

A	- area, square inches
B	- axial chord, inches
C <sub>L</sub>	- Zwiefel load coefficient
C <sub>L*</sub>	- load coefficient, $\Delta C_U/U$
E	- diffusion parameter
$\Delta H$	- work, Btu per pound
M	- Mach number
P	- pressure, psia
$\Delta P$	- pressure rise from minimum to exit value on suction surface
Q	- exit dynamic head
R <sub>C</sub>	- radius of curvature, inches
T	- temperature, °R
u	- tangential velocity, feet per second
X	- axial distance, inches
Y	- tangential distance, inches
Z	- number of airfoils
$\alpha$	- absolute gas angle, degrees
$\beta$	- relative gas angle degrees
$\tau$	- gap, inches

### Subscripts

0	inlet to first vane
1	inlet to first blade
2	exit from stage or airfoil section
G	gage point
S	static
T	total

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## SECTION I

### INTRODUCTION

(U) Future mission studies show that the bypass turbofan engine, which has seen increased use in military aircraft applications, will continue to be of primary interest. Characteristic of this type of engine is an increase in fuel economy with increased bypass ratio. In turn, high bypass ratio engines require higher fan power, which must be provided by the fan-drive or low-pressure turbine. This increased turbine power requirement must be met without a turbine efficiency penalty.

(U) Design analysis and optimization studies of aircraft jet engines involve a trade between component efficiencies and engine size and weight. The efficiency of a turbine is the result of optimization studies based on trade-offs between turbine diameter, rotational speed, number of stages and airfoil loading. The fan drive turbine design is constrained by further requirements. The rotational speed of the turbine must be limited in order that the fan tip Mach number does not exceed the limit for acceptable losses. At high by-pass ratios, where larger fan diameters are required, this problem is further aggravated. Applying conventional aerodynamics at fixed rotational speed, increased work can only be realized by further increases in the number of stages and/or the turbine diameter. Reduction of the turbine diameter or solidity results in a lighter turbine, but with a sacrifice in efficiency due to losses associated with increased loading. Considerable gains can be realized by an engine if the size and weight reduction can be made with no loss in efficiency. Therefore, current turbine technology must be improved in order to provide the desired level of performance for future turbofan engines.

(U) The objective of the work done under this contract is to analyze and test concepts which will increase the low pressure turbine loading while maintaining or increasing the turbine efficiency levels. The goals of this program are to develop turbine aerodynamic techniques and design procedures for efficient, high work, low-pressure turbines by means of analytical studies and cascade testing, and to demonstrate the effectiveness of the techniques by designing and testing a two-stage turbine that meets or exceeds the contract stage work and efficiency goals (See Section II).

(U) The total program is being conducted in four phases over a three year period, which commenced on 1 January 1968. Phase I defined the basic turbine design and an analysis of promising increased loading concepts was completed. The results of the Phase I study were reported in Reference 1. Phases II and III consisted of experimental testing to verify and extend the turbine aerodynamic techniques and design procedures for high loading levels. The design details of the baseline cascade test airfoils for both the annular segment and plane cascade rig were reported in Reference 2. The results of the annular cascade testing of the four baseline airfoils were reported in Reference 3. The annular cascade results of the baseline airfoil boundary layer control methods were reported in Reference 4. The results of the plane-cascade two-dimensional baseline investigation, the decreased-solidity annular cascade airfoil test results, and also the Phase III boundary layer control optimization and off-design evaluation are presented in this Report. Phase IV, consisting of the design and test of a two-stage demonstrator turbine, is currently in progress.

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(U) Work on the Contract during this report period proceeded to the conclusion of Phase II, including plane cascade tests of the baseline airfoils and annular segment cascade test of medium solidity airfoils. Phase III testing of the boundary layer control optimization and their off-design evaluation was also completed.

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## SECTION II

### BACKGROUND

(U) The High Stage Loading Low Pressure Turbine Contract consists of four phases. The testing and data analysis required in the first three phases of this Contract have been completed. Work on the Contract is currently proceeding in Phase IV.

(U) The results of the Phase I analysis are described in Reference 1. The objective of the Phase I study was to select a preliminary turbine design that is capable of meeting the performance requirements of this Contract (See Table I).

TABLE I  
TURBINE DESIGN PARAMETERS

Number of Stages	2
Average Load Coefficient, $C_L^*$	2.2
First Blade Tip Wheel Speed	1000 fps
First Blade Inlet Hub-Tip Diameter Ratio	$\leq .8$
Exit Swirl Angle - Without Exit Guide Vanes	$\leq 20^\circ$
Exit Swirl Angle - With Exit Guide Vanes	$0^\circ$
Turbine Inlet Temperature	1450°F
Airflow	$\geq 50$ lbs/sec
Average Stage Efficiency	91%
Life	10,000 hrs

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(U) As part of Phase I, the analyses included the consideration of flowpath, reaction level, load coefficient level and variations in work distribution for which velocity triangles were generated. Furthermore, preliminary airfoil contours were defined for the same velocity triangles at three levels of reaction, and three levels of solidity for the resulting stages.

(U) Based on the Phase I study, and after additional airfoil refinement, the medium-solidity airfoil turbine design was selected for the Phase II and III evaluation. Phase II consisted of two- and three-dimensional experimental evaluation of the four baseline airfoils, investigation of promising end-wall boundary layer control techniques and evaluation of the baseline airfoils at a 15 percent higher Zweifel loading coefficient. The results of the two-dimensional evaluation are presented in this Report. The partial results of the medium solidity evaluation were presented in Reference 4, and are concluded in this Report. The results of the three-dimensional baseline airfoil evaluation were presented in Reference 3.

(U) The second vane was chosen for the end wall loss study by the process of elimination. The first stage vane and blade inside diameter and wall extensions indicated that separation occurred close enough to the test cascade to influence probe readings near this wall in future tests if it moved upstream. The short second stage blade chord could have made the fabrication of end wall boundary layer control techniques more difficult. The second vane, therefore, was the vehicle for four boundary layer control techniques, including the use of flow fence

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and increased surface roughness, local airfoil recontouring, end wall contouring, and local airfoil recambering. The results of these evaluations (published in Reference 4) indicated that within the limited range of investigation of each technique, local recambering was the most effective.

(U) Local recambering was further pursued during Phase III evaluations, where three additional recambered designs were investigated. These were evaluated at both design and off-design conditions and the results of these tests are reported herein.

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## **SECTION III**

### **TWO-DIMENSIONAL DESIGN VERIFICATION (TASK IIa)**

#### **1. RFP OBJECTIVE**

(U) Provide an experimental verification of two-dimensional design characteristics.

#### **2. TASK OBJECTIVE**

(U) The purpose of this Task was to conduct plane cascade tests in order to verify the aerodynamic concepts applied to the turbine design during the Phase I Program, and to establish the two-dimensional loss levels for the chosen turbine airfoil profiles at design conditions.

(U) The plane cascade tests served two equally important purposes. First, the measured profile losses were compared with those contained in the existing design procedures in order to verify their accuracy. Secondly, the total pressure and flow angle profiles at the exit plane indicated whether or not the surface boundary layer had separated. Each airfoil was designed so that such two-dimensional separation should not have occurred, and these tests constituted a verification of the entire airfoil section design procedure.

#### **3. CASCADE PACK AND FACILITY DESIGN**

(U) A summary of the pertinent design parameters for the airfoils tested in the plane cascade rig is shown in Table II. The six airfoil sections which were investigated are the mean section of each airfoil row and the second vane and second blade root sections. The cascade packs consisted of straight airfoils having the contours of their respective baseline airfoil designs. These airfoil sections were shown in Reference 2, Section II. In this same report, the airfoil coordinates, scale factors, and locations of airfoil static pressure taps were also presented. In addition, the method of assembly of the test packs and locations of the exit static taps on each test pack were also presented.

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TABLE II

## MEDIUM-REACTION NORMAL-SOLIDITY AIRFOIL SUMMARY

<u>First Stage Vane</u>	<u>No. of Foils</u>	<u>Exit Mach No.</u>	<u>ΔP/Q Design</u>	<u>Max. Surface Mach No.</u>
Root	62	0.949	0.313	1.150
¼ Root		0.891	0.244	1.036
Mean		0.852	0.213	0.966
¼ Tip		0.820	0.195	0.918
Tip		0.770	0.192	0.859
<u>First Stage Blade</u>				
Root	116	0.890	0.279	1.058
¼ Root		0.849	0.296	1.020
Mean		0.788	0.331	0.968
¼ Tip		0.738	0.294	0.883
Tip		0.730	0.236	0.840
<u>Second Stage Vane</u>				
Root	80	1.003	0.337	1.230
¼ Root		0.920	0.277	1.094
Mean		0.870	0.227	0.999
¼ Tip		0.821	0.196	0.928
Tip		0.765	0.203	0.865
<u>Second Stage Blade</u>				
Root	126	0.944	0.408	1.233
¼ Root		0.922	0.324	1.136
Mean		0.904	0.263	1.067
¼ Tip		0.897	0.236	UNCLASSIFIED 1.040
Tip		0.892	0.224	1.025

(U) The design of the plane cascade is shown schematically in Figure 1. The test section had a variable inlet and exit flow path in order that all six cascades could be evaluated using basically the same hardware. The variable geometry rig design permitted each cascade to be tested at its respective design incidence angle. One change was made to the rig (previously shown in Reference 2, Figure 15) after the initial rig checkout run was made. This included the addition of turning vanes in the inlet channel, since, as initially designed, there was a small flow distortion at the inlet to the test cascade pack. The problem was corrected by this change.

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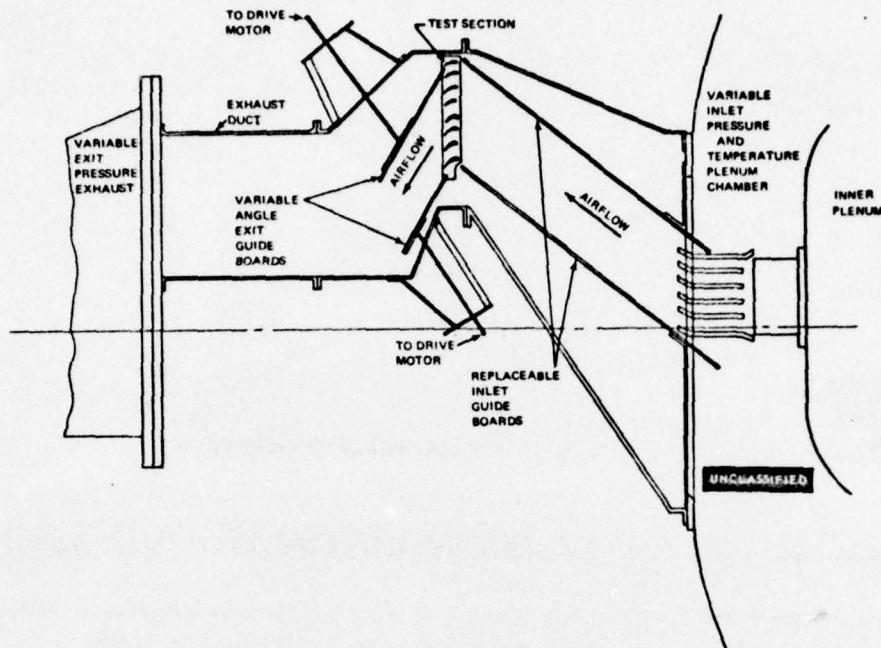


Figure 1 Schematic of Revised Plane Cascade Rig

(U) During the testing, both inlet and exit pressures were independently controlled, as well as the inlet temperature level, allowing the correct simulation of inlet and exit Mach numbers, and Reynolds number. The cascade pack total pressure and exit flow angle profiles were measured. Local and gapwise integrated loss coefficients were calculated for the test channels.

## 4. DISCUSSION

(U) A comparison of the loss coefficient data for the four plane cascade mean sections with the midspan values measured in the four normal solidity part annular cascade is presented in Table III. In addition, a comparison of these data with losses predicted by two different methods is shown. One of the predicted values was calculated using a modified Ainley and Mathieson correlation based on global airfoil design parameters. The other prediction was calculated using fully turbulent boundary layer characteristics computed by the Airfoil Boundary Layer Deck, which utilized the predicted airfoil surface static pressure distributions. The airfoil static pressure distributions were calculated by the Airfoil Pressure Distribution Program. It can be seen from this comparison (Table III) that the plane cascade loss levels are consistent with the midspan values from the annular segment cascade, and both are consistently below either of the predicted values for each airfoil. This is probably due to an extended region of laminar boundary resulting from low cascade inlet turbulence levels. The screen results uphold this judgement but may not yet reproduce the loss levels that will be obtained in the rotating machine.

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TABLE III

MEASURED PLANE CASCADE LOSS COMPARISON WITH  
MEASURED MIDSPAN ANNULAR CASCADE LOSS AND  
PREDICTED LOSS AT DESIGN REYNOLDS NUMBERS

		Loss Coefficient Measured, $1-\phi^2$		Loss Coefficient Predicted, $1-\phi^2$	
Midspan Section	Exit Mach Number	Plane Cascade	Annular Cascade	Global Correlation	Fully Turbulent Boundary Layer
First Vane	0.852	0.025	0.017 0.023*	0.031	0.031
First Blade	0.788	0.025	0.027	0.036	0.049
Second Vane	0.870	0.024	0.021 0.028*	0.036	0.034
Second Blade	0.904	0.023	0.028	0.030	0.040

\*With inlet turbulence screen.

(U) Table IV shows the measured loss levels for the four midspan sections as compared with the predicted loss levels using the Airfoil Boundary Layer Deck. In the boundary layer calculations previously mentioned (Table III), fully turbulent flow was assumed and transition was predicted internally by the program. For the values shown in Table IV, transition was fixed at the last minimum pressure. The excellent agreement shown here, in the three cases out of four that converged without separating, makes it appear that the selected transition point is correct. Additional proof is offered by the data.

TABLE IV  
MEASURED PLANE CASCADE LOSSES COMPARED TO  
PREDICTED VALUES USING THE BOUNDARY LAYER DECK  
WITH TRANSITION ASSUMED AT THE SECOND MINIMUM  
PRESSURE

Midspan Section	Measured Loss Coefficient $1-\phi^2$	Predicted Loss Coefficient $1-\phi^2$
First Vane	0.025	0.025
First Blade	0.025	0.026
Second Vane	0.024	0.022
Second Blade	0.023	Separation Indicated

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(U) The complete set of loss and measured exit flow angle data at various Reynolds numbers is presented as a function of Mach number in Figures 2 through 13. The lack of variation of the loss coefficient with Reynolds number at constant Mach number indicates that the transition point was stationary during the tests. The loss coefficient curves indicate a strong transonic drag rise on each airfoil section.

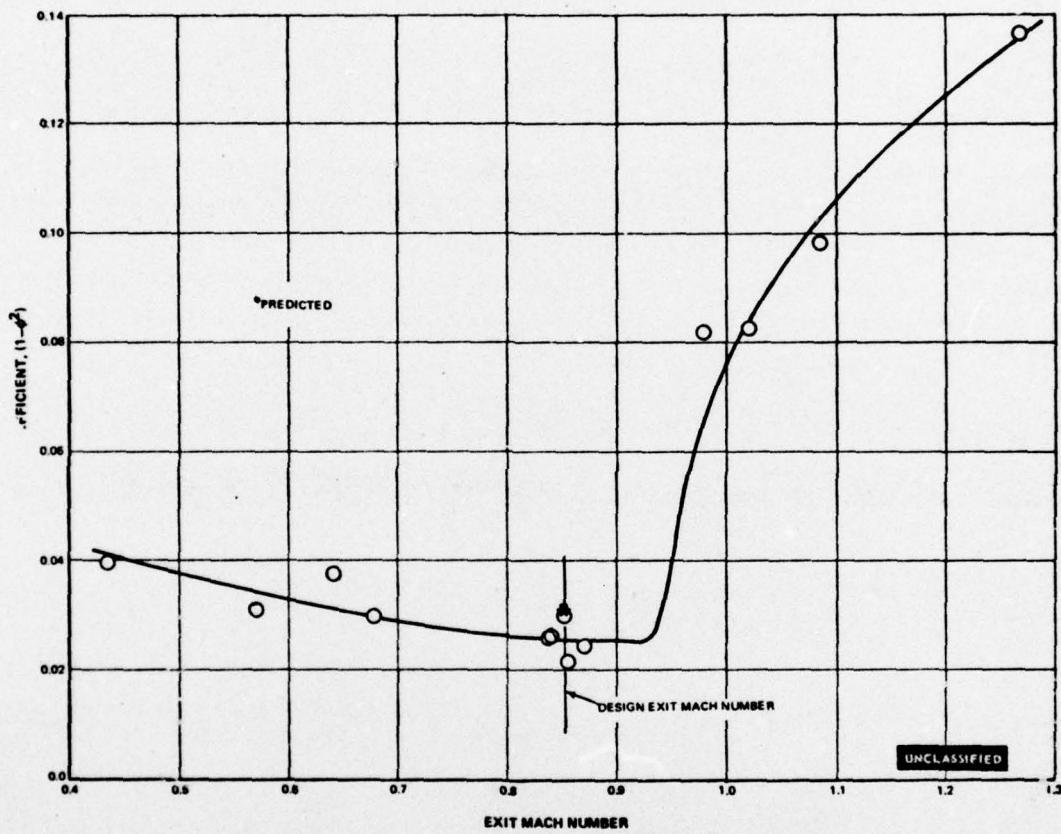


Figure 2 Variation of Loss Coefficient With Exit Mach Number - First Vane Mean Section

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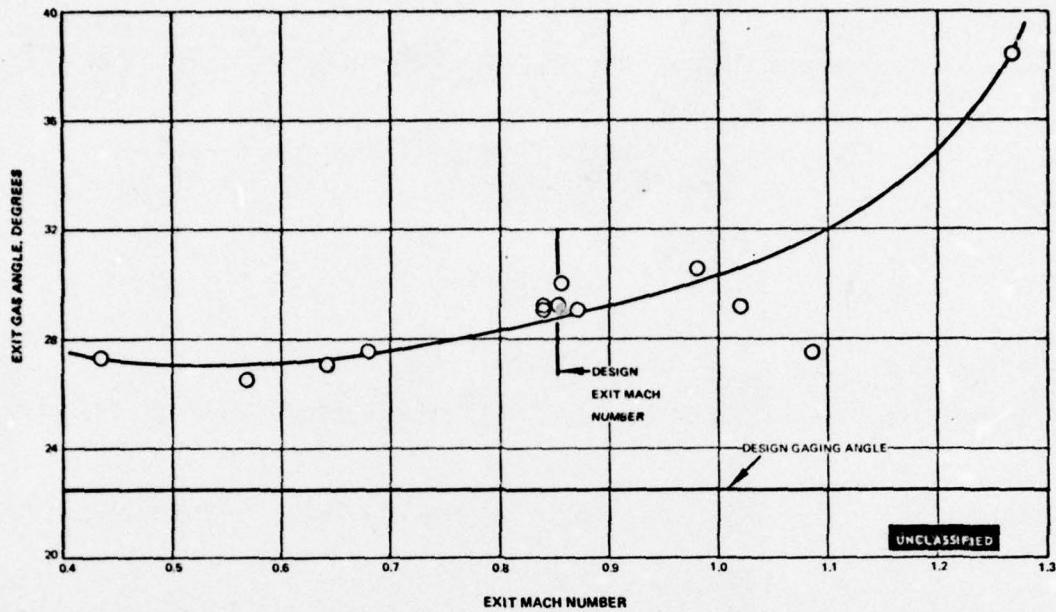


Figure 3 Variation of Exit Gas Angle with Exit Mach Number - First Vane Mean Section

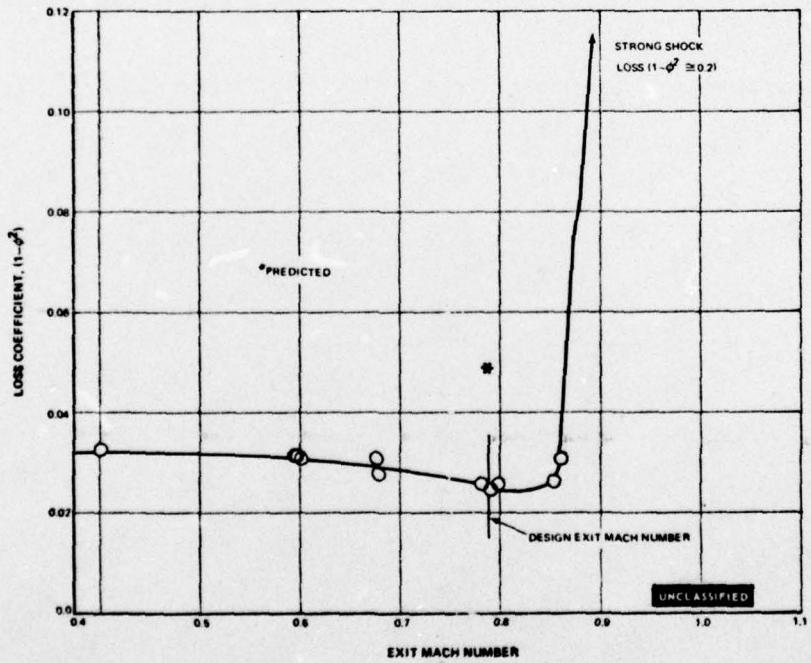


Figure 4 Variation of Loss Coefficient with Exit Mach Number - First Blade Mean Section

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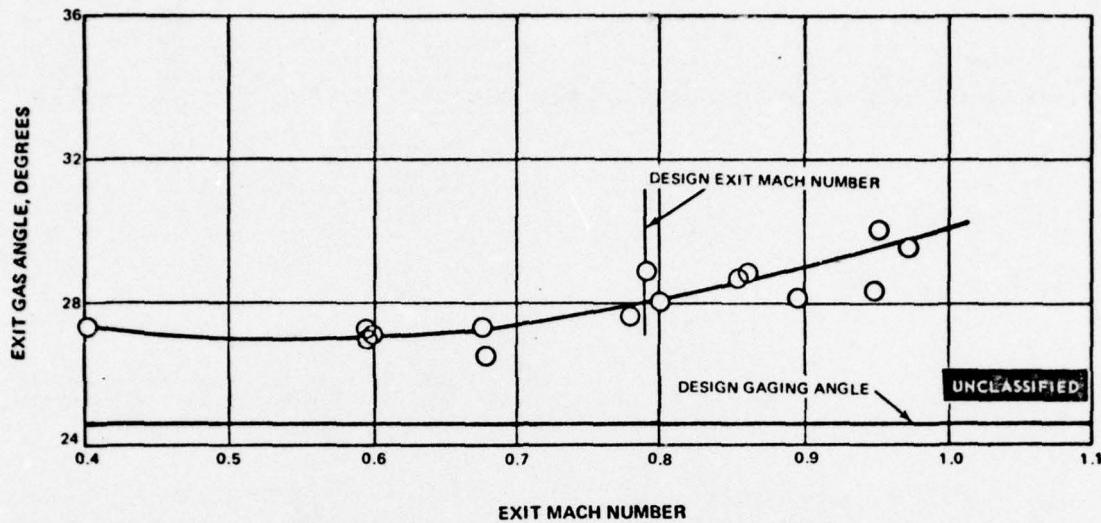


Figure 5 Variation of Exit Gas Angle with Exit Mach Number - First Blade Mean Section

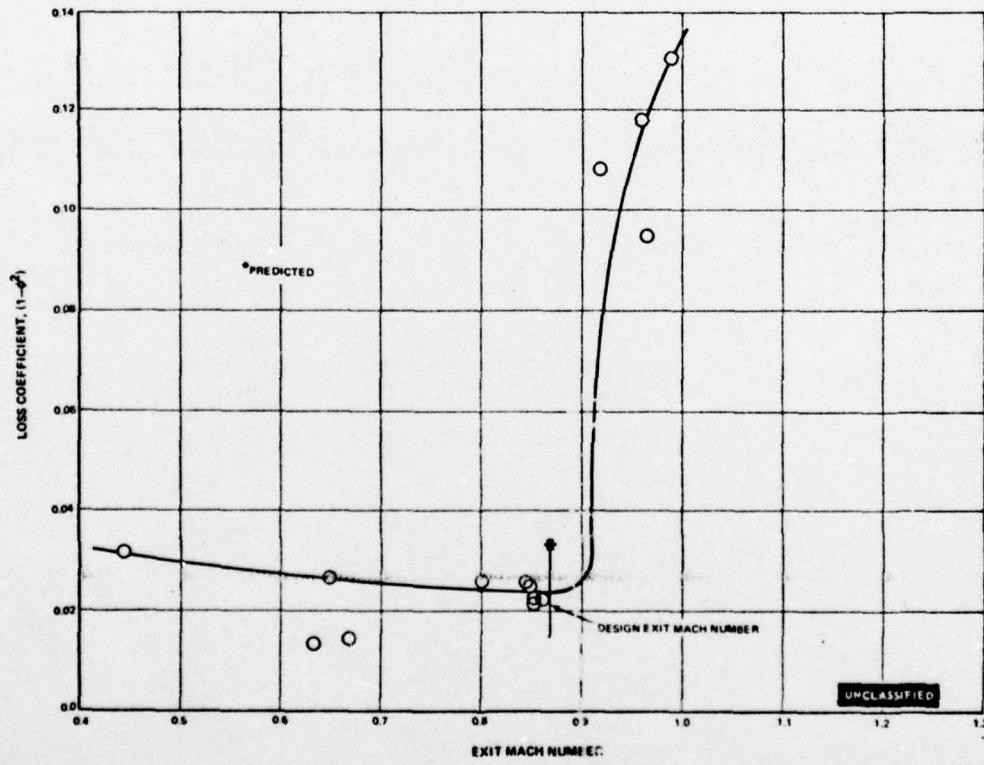


Figure 6 Variation of Loss Coefficient with Exit Mach Number - Second Vane Mean Section

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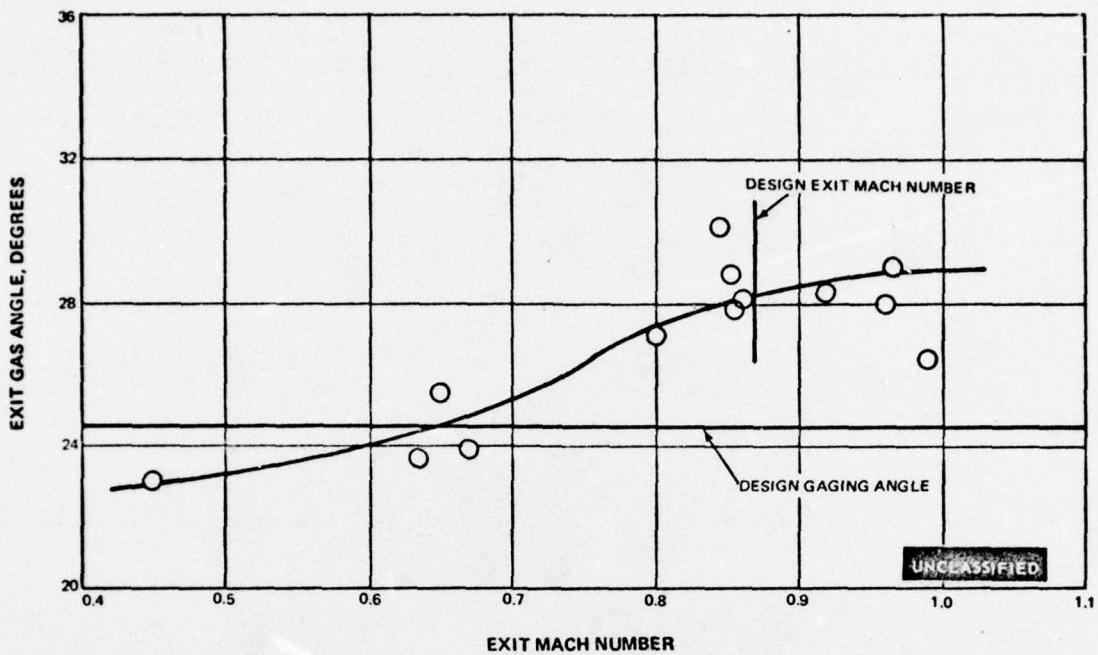


Figure 7 Variation of Exit Gas Angle with Exit Mach Number - Second Vane Mean Section

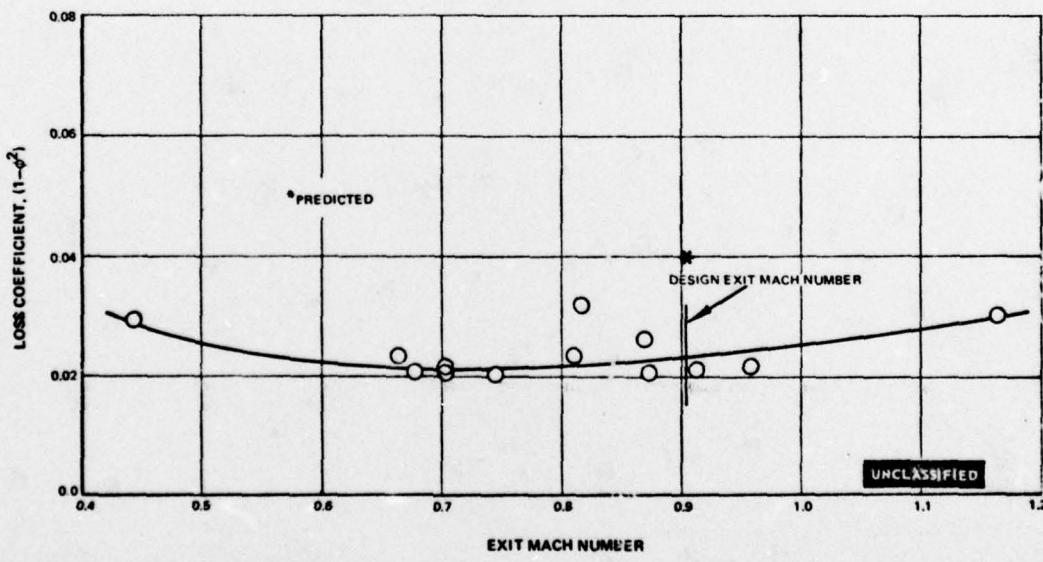


Figure 8 Variation of Loss Coefficient with Exit Mach Number - Second Blade Mean Section

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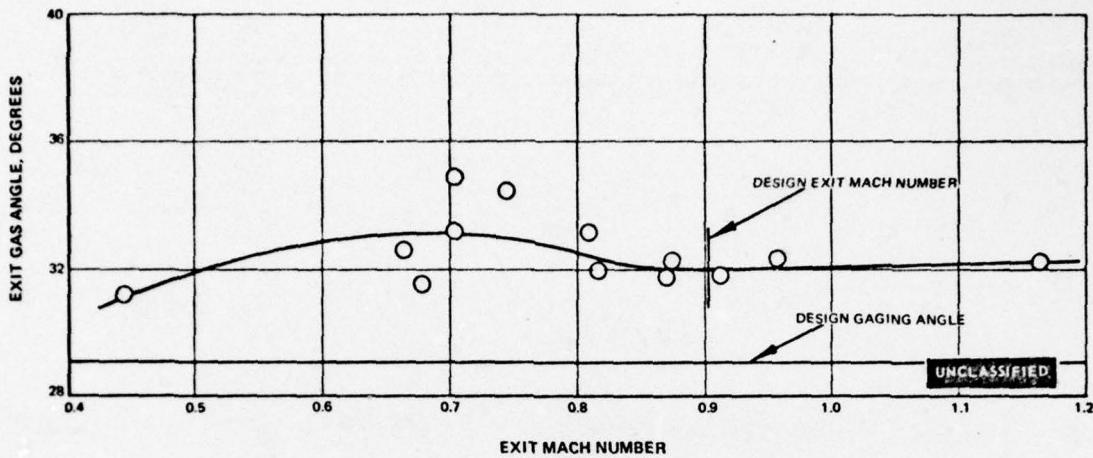


Figure 9 Variation of Exit Gas Angle with Exit Mach Number - Second Blade Mean Section

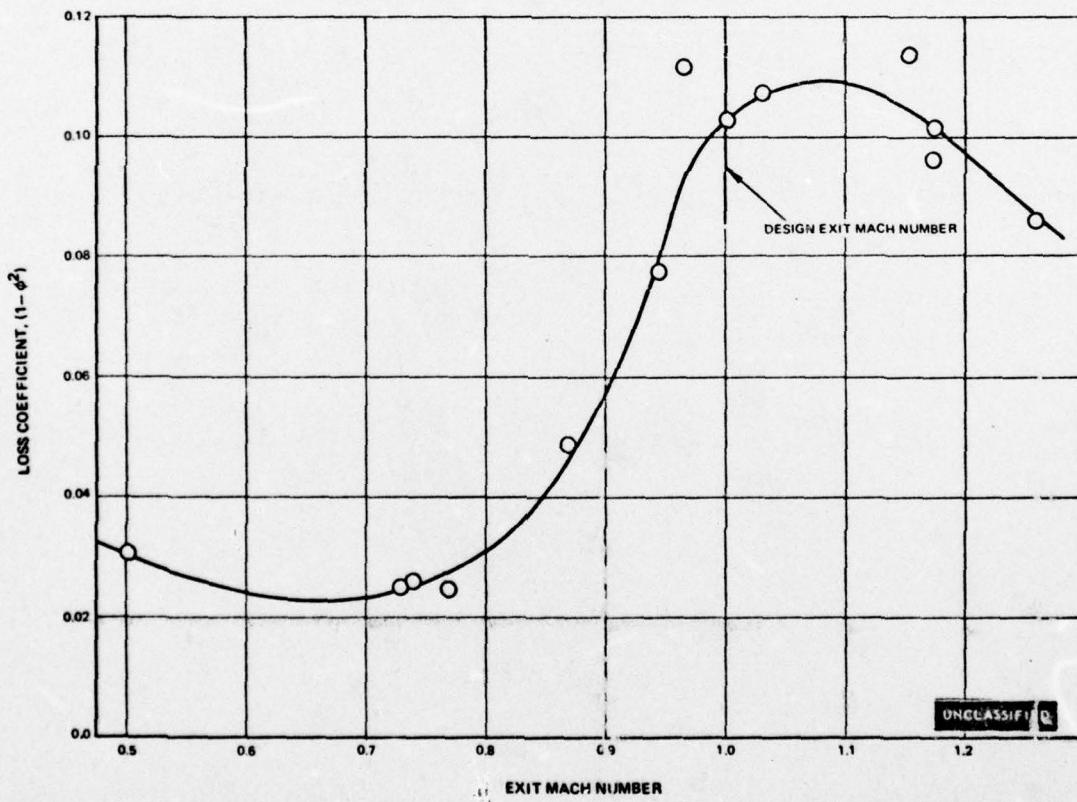


Figure 10 Variation of Loss Coefficient with Exit Mach Number - Second Vane Root Section

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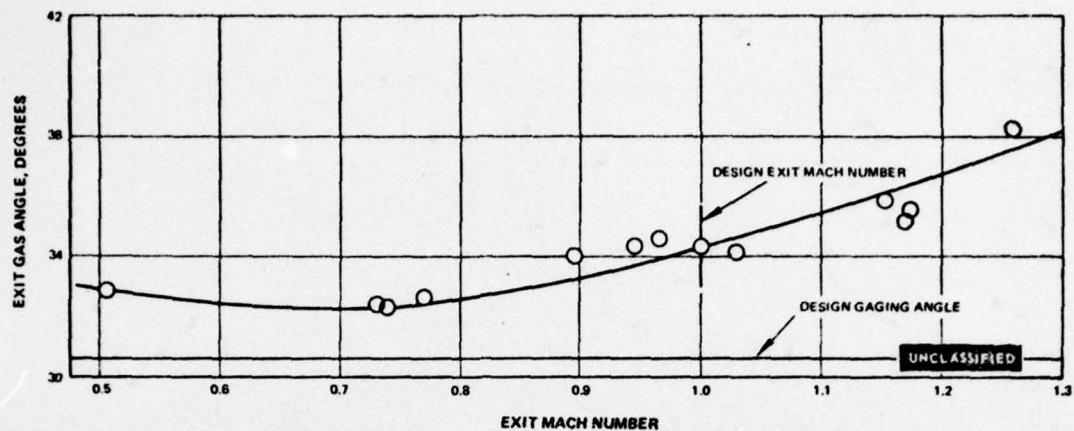


Figure 11 Variation of Exit Gas Angle with Exit Mach Number - Second Vane Root Section

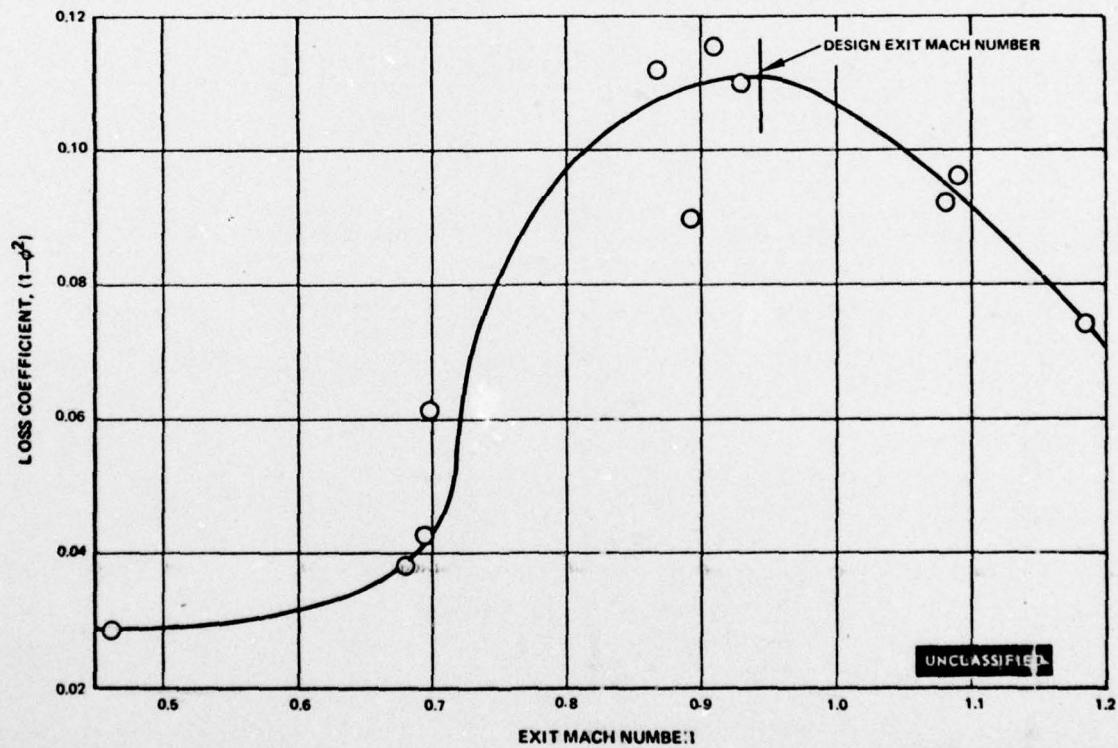


Figure 12 Variation of Loss Coefficient with Exit Mach Number - Second Blade Root Section

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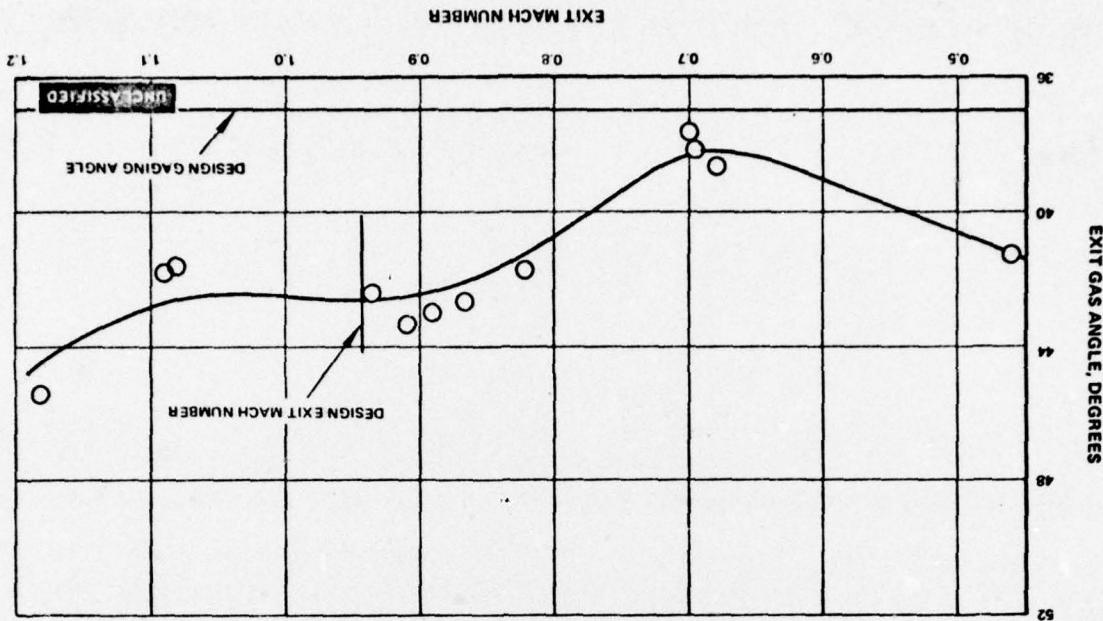


Figure 13 Variation of Exit Gas Angle with Exit Mach Number - Second Blade Root Section

(U) Figures 14 through 39 show the measured and predicted airfoil surface pressure distributions. The four midspan sections performed as expected in the vicinity of the design point. The two root sections, however, show a transonic drag rise before the design point is reached. In view of this, all of the six sections were reevaluated at their respective design points. The pressure distributions that would have been predicted using the new Transonic Pressure Distribution Deck, which is being used in the final contract turbine design, are shown for the six airfoils in Figures 16, 20, 24, 29, 34 and 37. In these figures, the pressure distributions predicted by the transonic pressure distribution deck and shown for the four mean sections, namely figures 16, 20, 24 and 29 are in good agreement with both the data and the original Airfoil Pressure Distribution Program. However, for the two root sections, there is a large disparity between the original and the new predictions; the latter being considerably closer to the data. The root sections have low reaction levels and high exit Mach numbers. The Transonic Pressure Distribution Program has been shown to be very effective when used to predict mixed-flow problems of this nature. The principal changes that the new program predicts are: less loading in the forward portion of the airfoil and higher loading (over-acceleration) in the rearward portion of the foil. The result of this is a shock, a condition necessary to get the over-accelerated flow up to the operating back-pressure, which stands off of the trailing edge of both airfoils. The measured pressure distributions indicate this by the very sharp pressure rises shown between 90 and 100 percent chord for both root sections. Using the Transonic Pressure Distribution Deck, then, airfoils may be designed to avoid this problem. The final design will utilize this advanced design technique.

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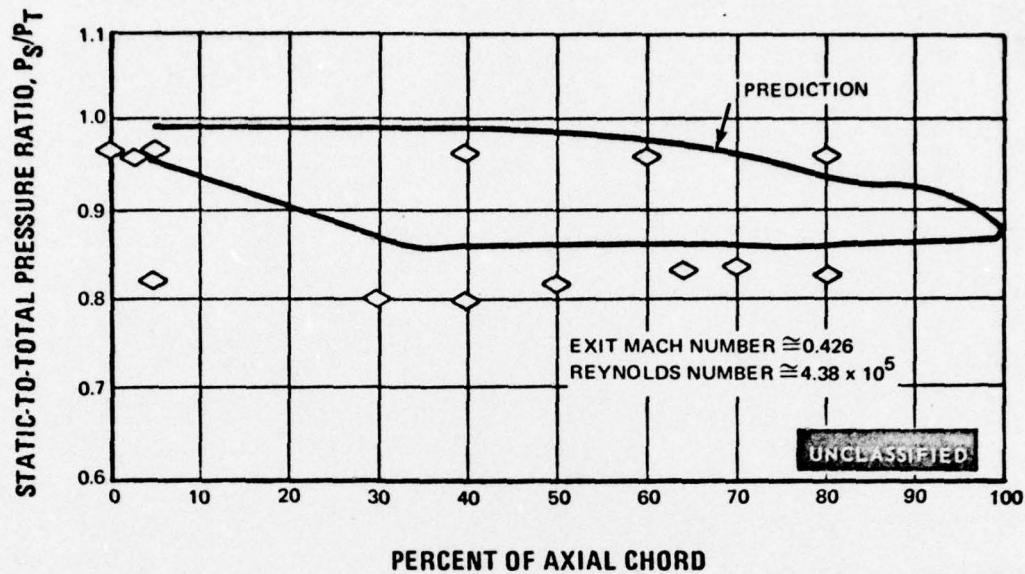


Figure 14 Static-to-Total Pressure Ratio Versus Percent of Axial Chord, First Vane Mean Section; Exit Mach Number = .426

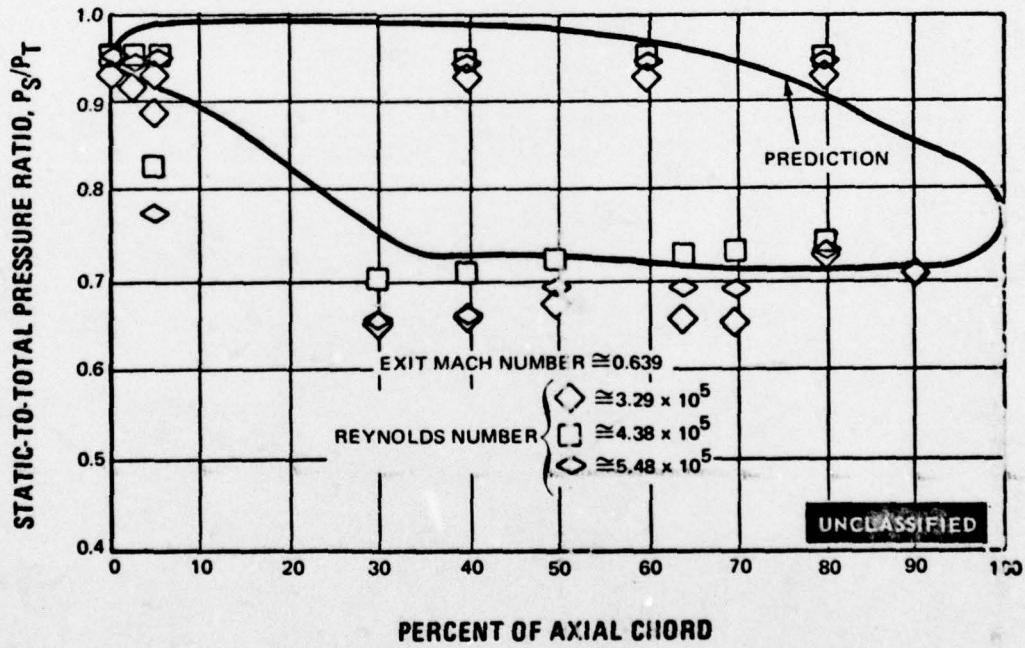


Figure 15 Static-to-Total Pressure Ratio Versus Percent of Axial Chord, First Vane Mean Section; Exit Mach Number ≈ .639

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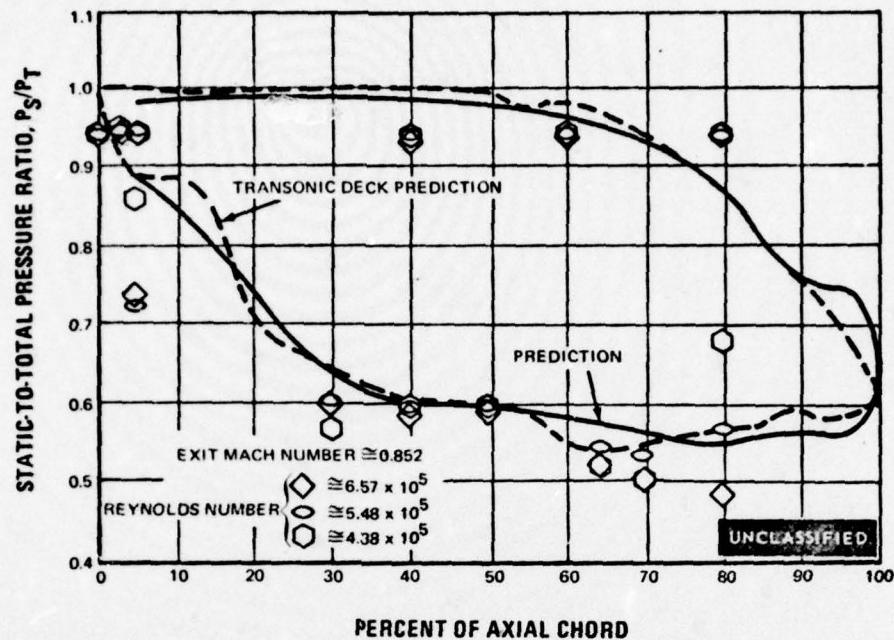


Figure 16 Static-to-Total Pressure Ratio Versus Percent of Axial Chord, First Vane Mean Section; Exit Mach Number = .852

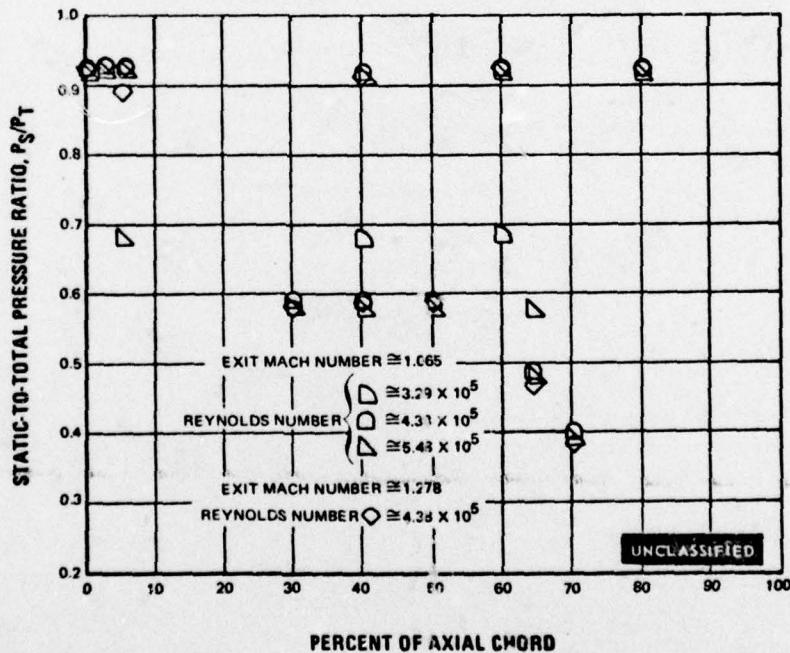


Figure 17 Static-to-Total Pressure Ratio Versus Percent of Axial Chord, First Vane Mean Section; Exit Mach Number = 1.065 and 1.278

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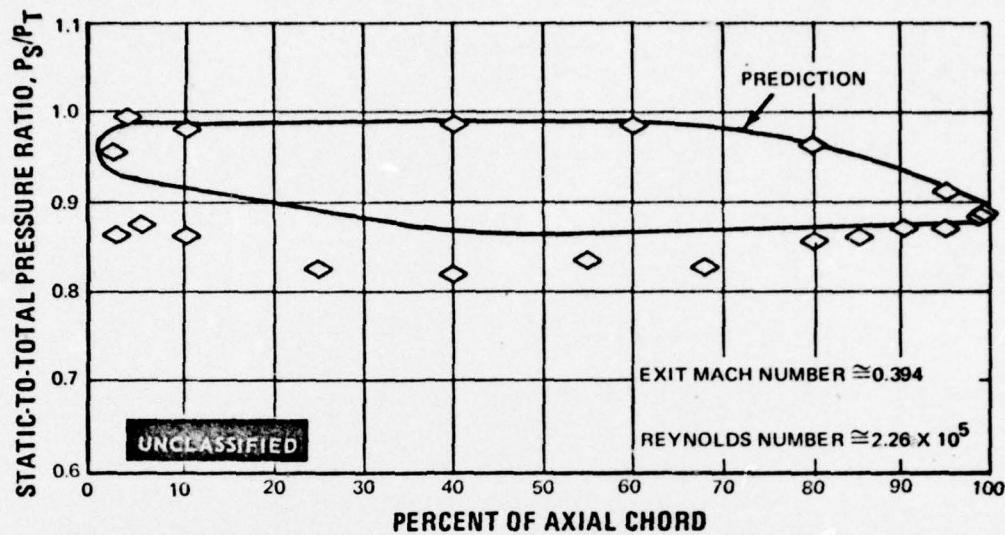


Figure 18    Static-to-Total Pressure Ratio Versus Percent of Axial Chord, First Blade Mean Section; Exit Mach Number = .394

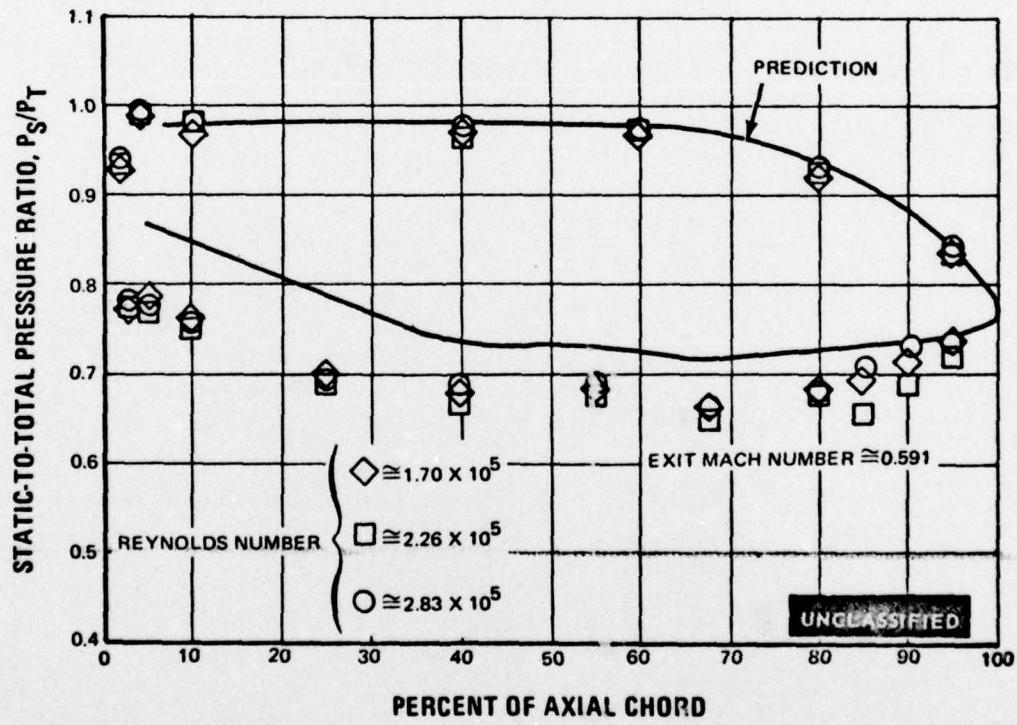


Figure 19    Static-to-Total Pressure Ratio Versus Percent of Axial Chord, First Blade Mean Section; Exit Mach Number = .591

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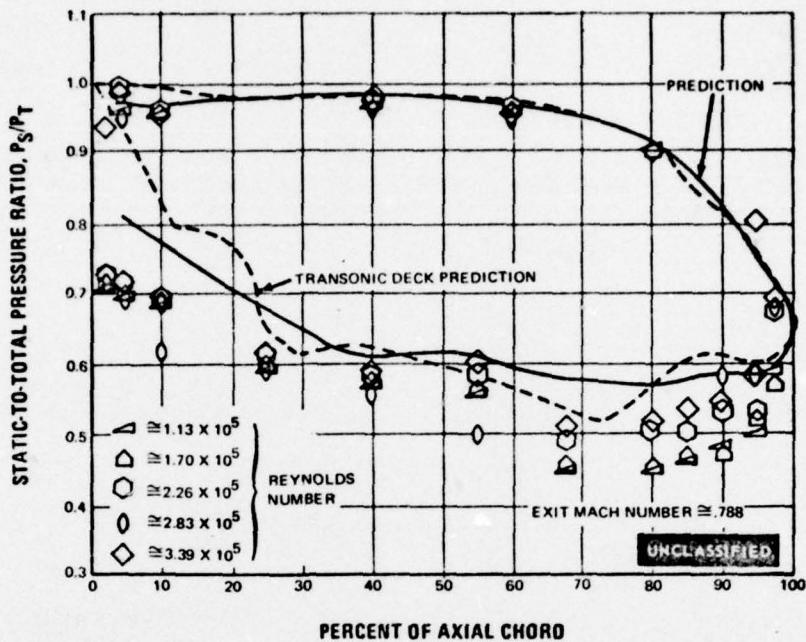


Figure 20 Static-to-Total Pressure Ratio Versus Percent of Axial Chord, First Blade Mean Section; Exit Mach Number = .788

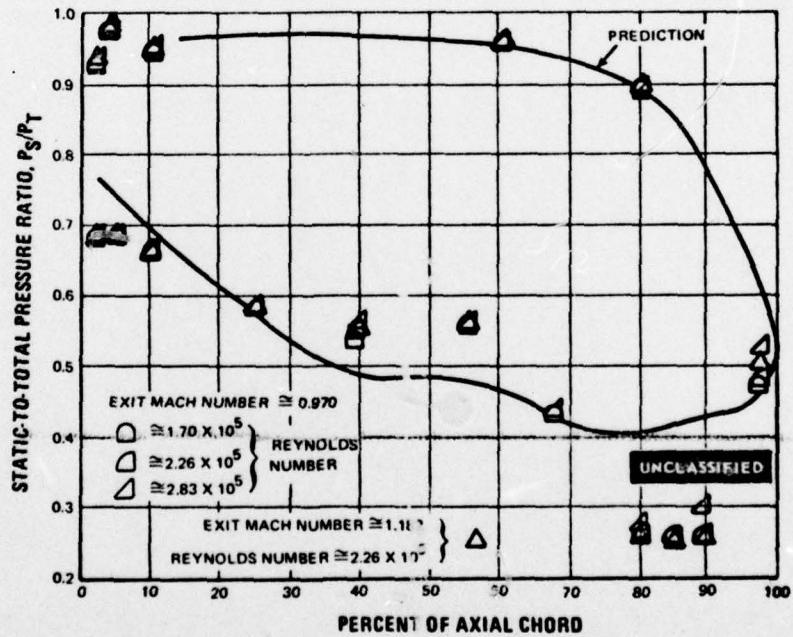


Figure 21 Static-to-Total Pressure Ratio Versus Percent of Axial Chord, First Blade Mean Section; Exit Mach Number = .970 and 1.182

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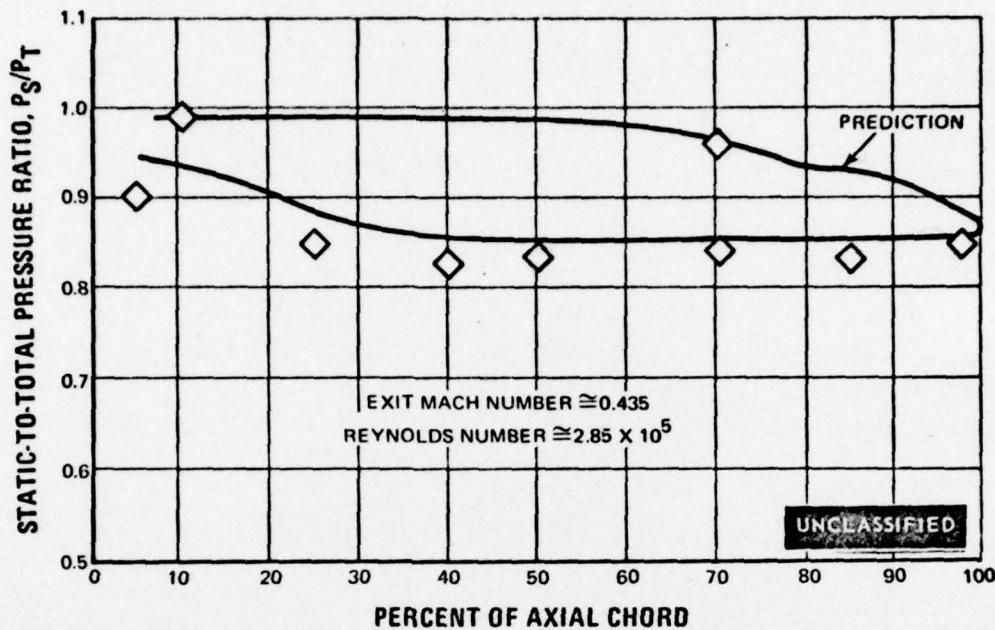


Figure 22 Static-to-Total Pressure Ratio Versus Percent of Axial Chord, Second Vane Mean Section; Exit Mach Number = .435

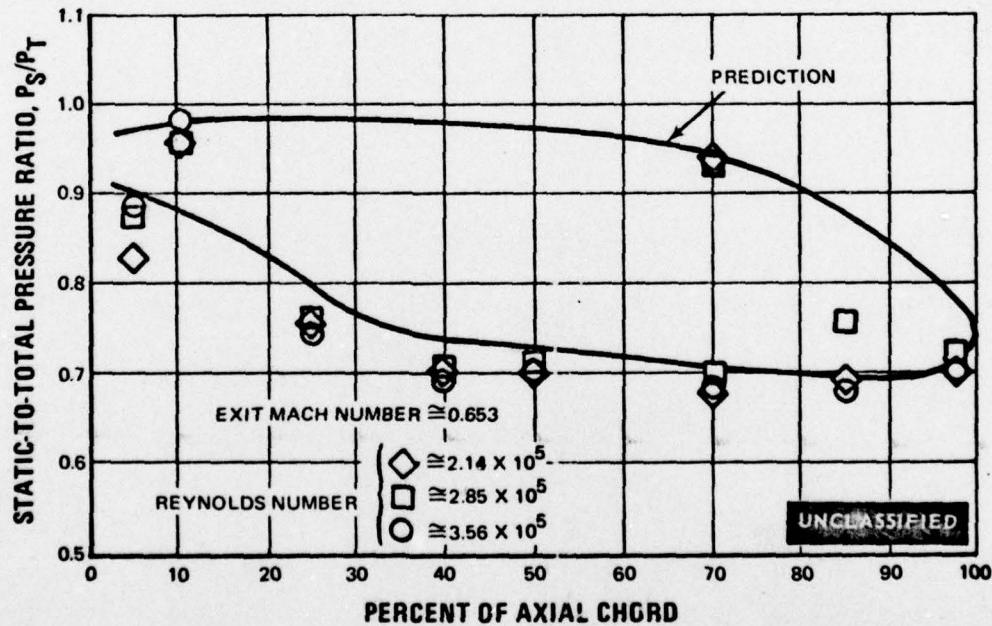


Figure 23 Static-to-Total Pressure Ratio Versus Percent of Axial Chord, Second Vane Mean Section; Exit Mach Number = .653

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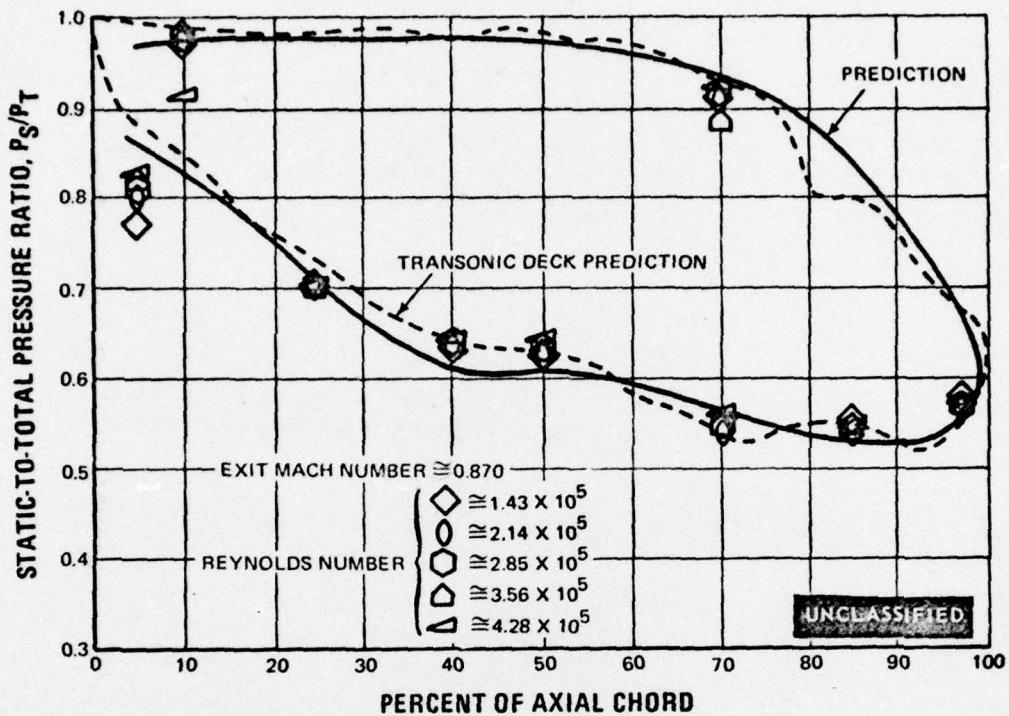


Figure 24 Static-to-Total Pressure Ratio Versus Percent of Axial Chord, Second Vane Mean Section; Exit Mach Number = .870

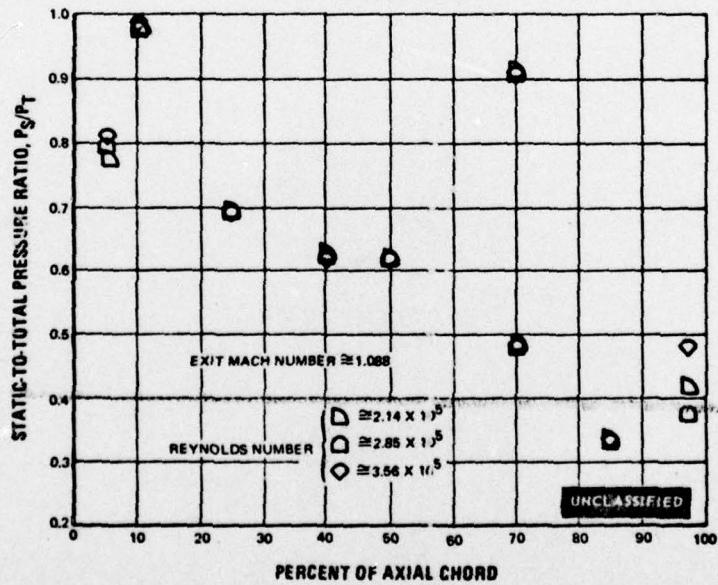


Figure 25 Static-to-Total Pressure Ratio Versus Percent of Axial Chord, Second Vane Mean Section; Exit Mach Number = 1.088

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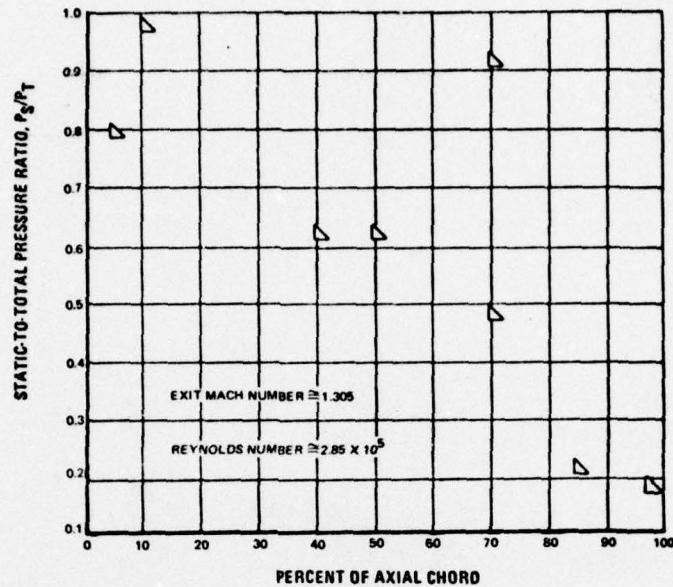


Figure 26 Static-to-Total Pressure Ratio Versus Percent of Axial Chord, Second Vane Mean Section; Exit Mach Number = 1.305

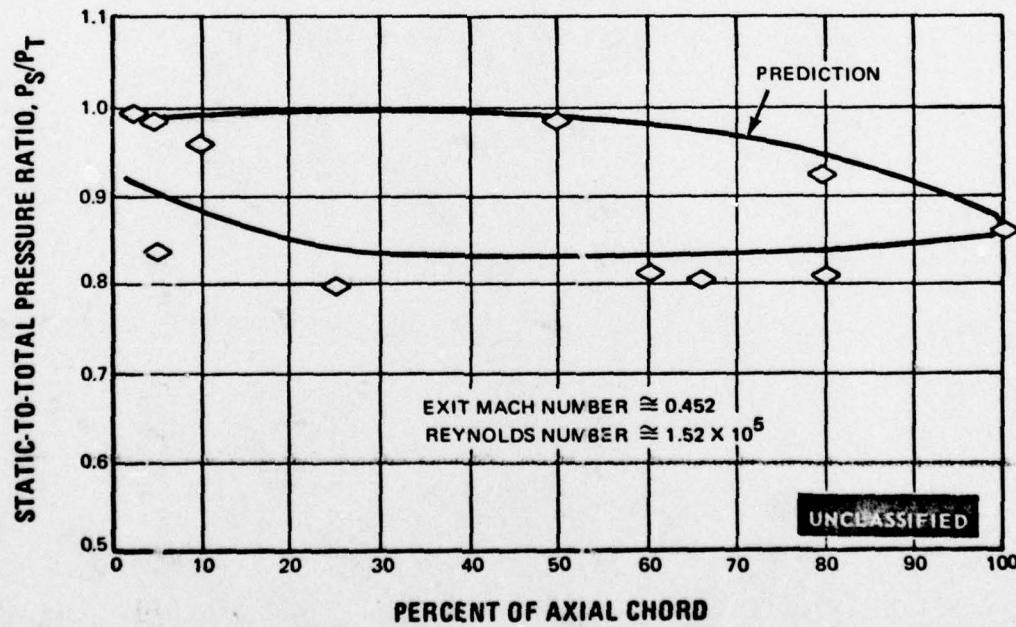


Figure 27 Static-to-Total Pressure Ratio Versus Percent of Axial Chord, Second Blade Mean Section; Exit Mach Number = .452

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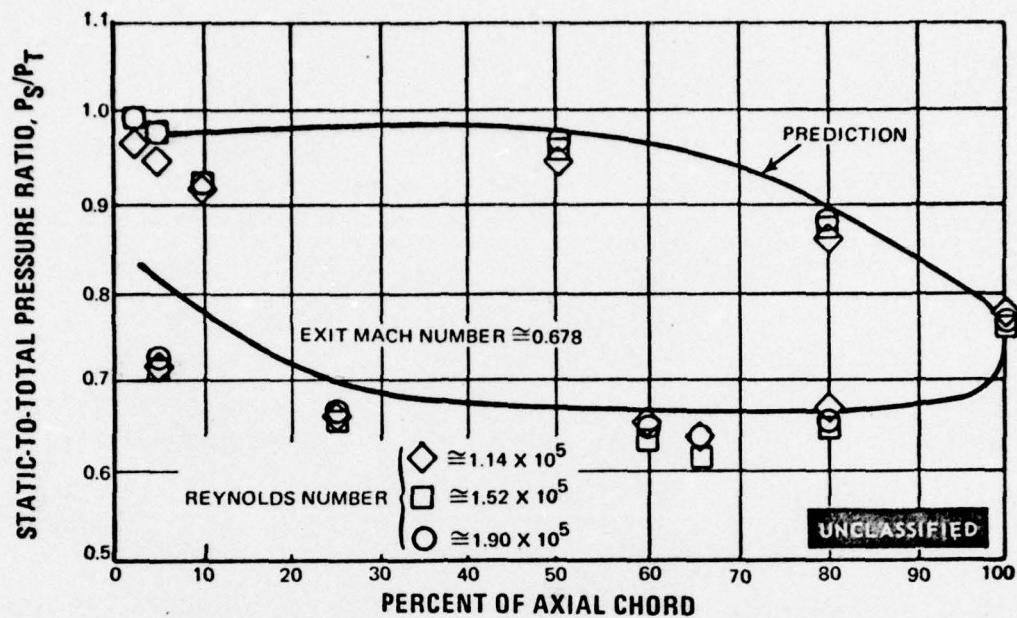


Figure 28 Static-to-Total Pressure Ratio Versus Percent of Axial Chord, Second Blade Mean Section; Exit Mach Number = .678

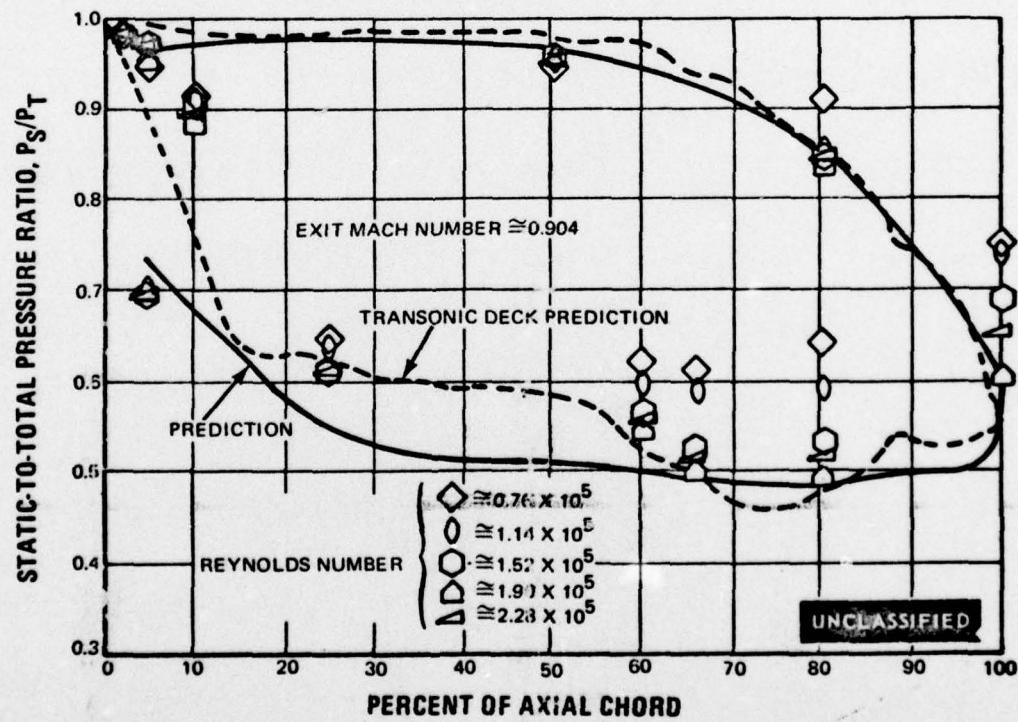


Figure 29 Static-to-Total Pressure Ratio Versus Percent of Axial Chord, Second Blade Mean Section; Exit Mach Number ≈ .904

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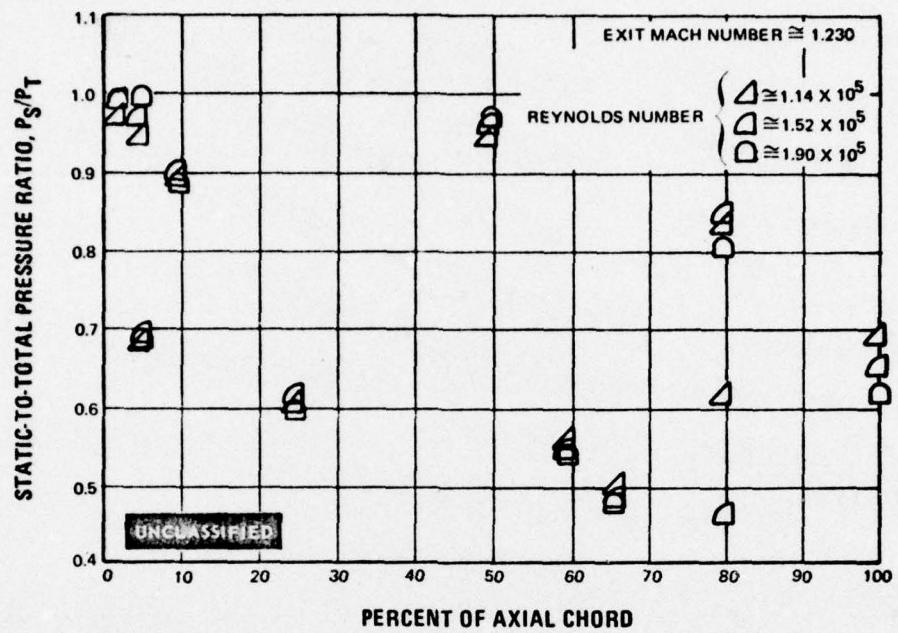


Figure 30 Static-to-Total Pressure Ratio Versus Percent of Axial Chord, Second Blade Mean Section; Exit Mach Number = 1.230

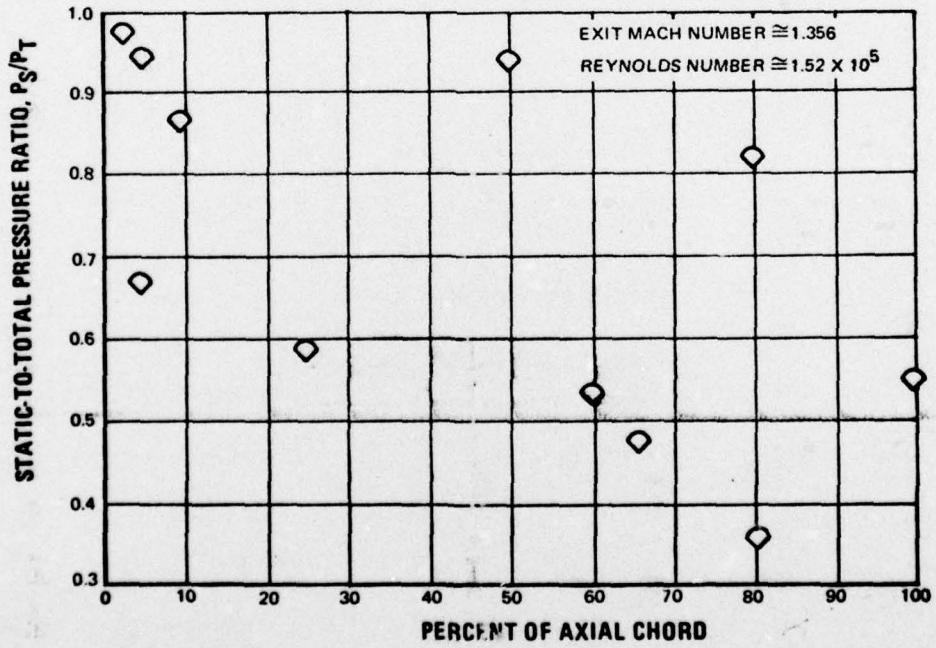


Figure 31 Static-to-Total Pressure Ratio Versus Percent of Axial Chord, Second Blade Mean Section; Exit Mach Number = 1.356

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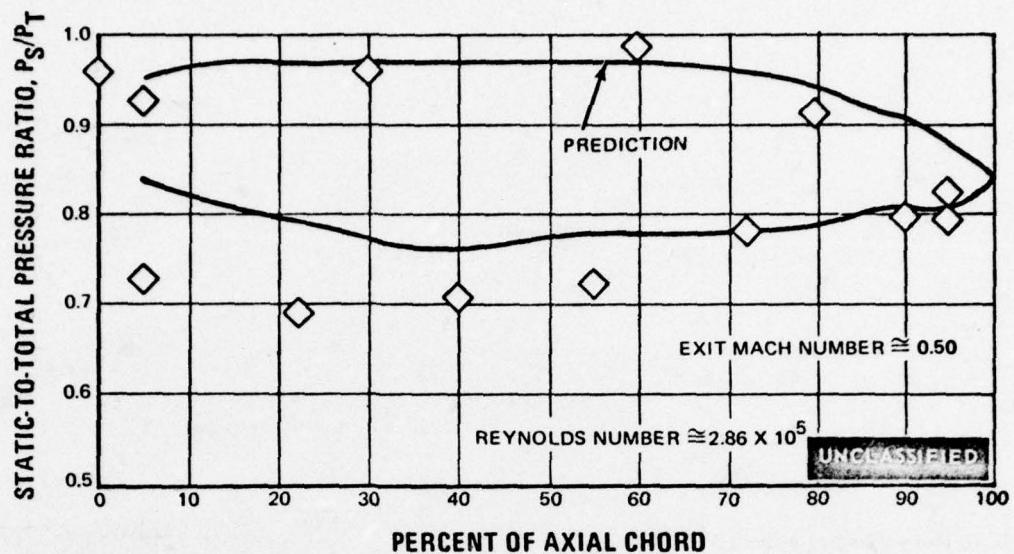


Figure 32 Static-to-Total Pressure Ratio Versus Percent of Axial Chord, Second Vane Root Section; Exit Mach Number = .50

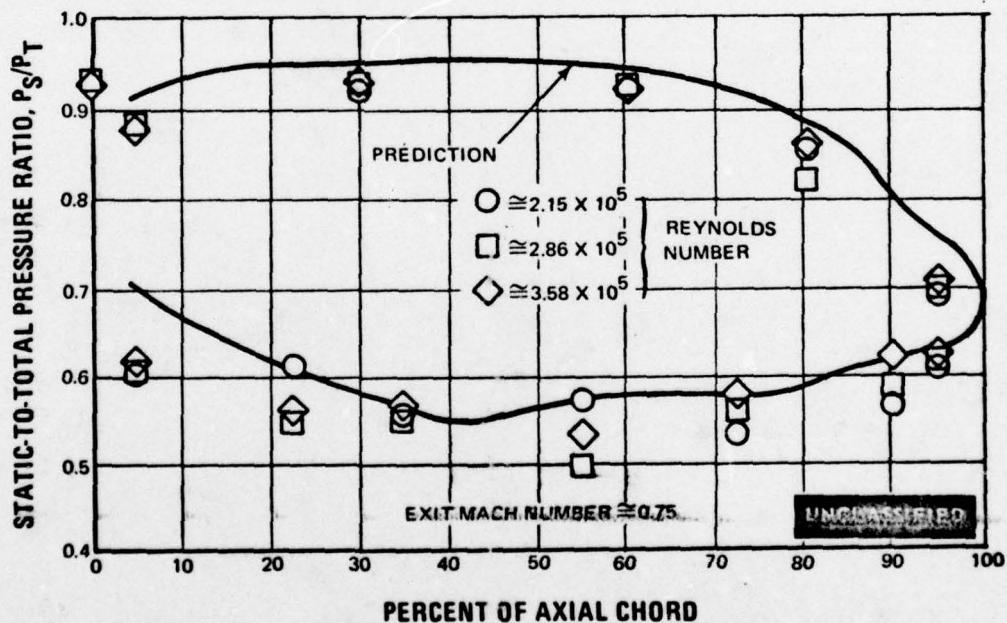


Figure 33 Static-to-Total Pressure Ratio Versus Percent of Axial Chord, Second Vane Root Section; Exit Mach Number = .75

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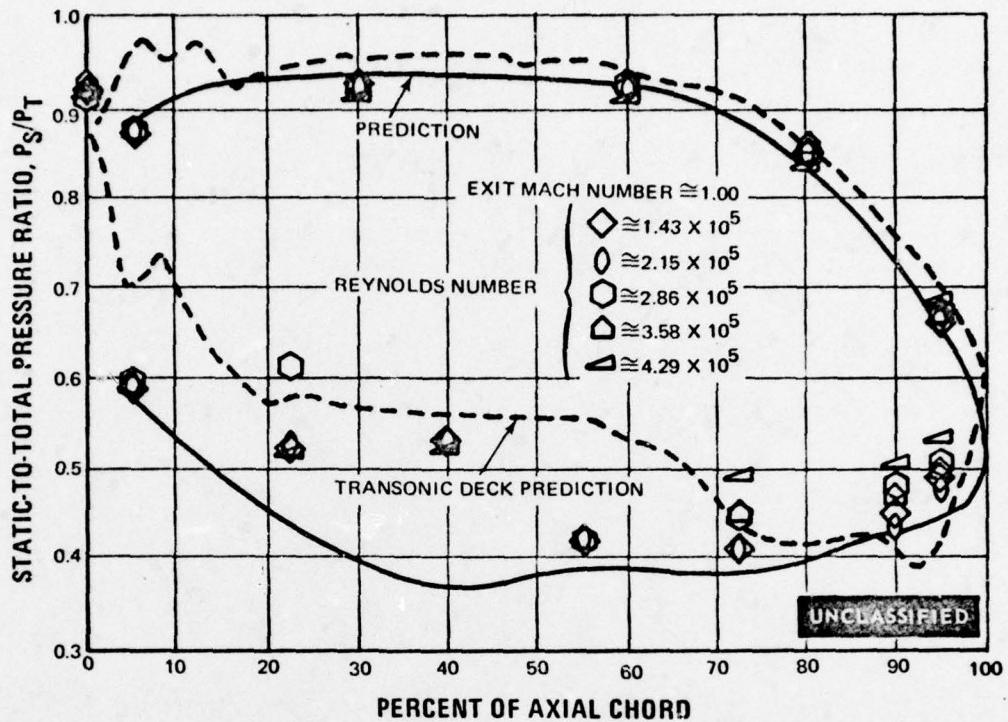


Figure 34 Static-to-Total Pressure Ratio Versus Percent of Axial Chord, Second Vane Root Section; Exit Mach Number = 1.00

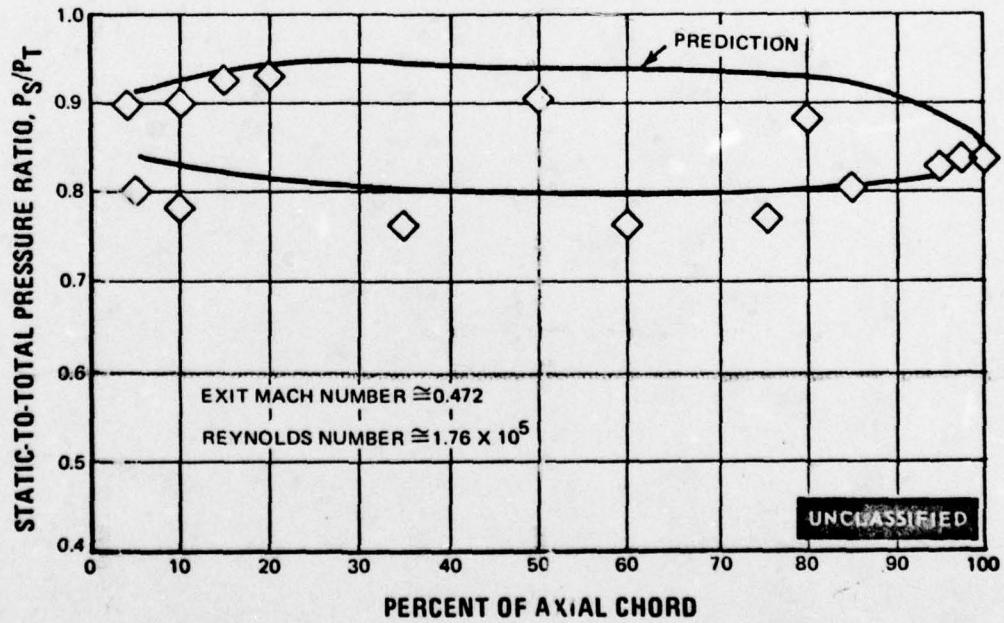


Figure 35 Static-to-Total Pressure Ratio Versus Percent of Axial Chord, Second Blade Root Section; Exit Mach Number = .472

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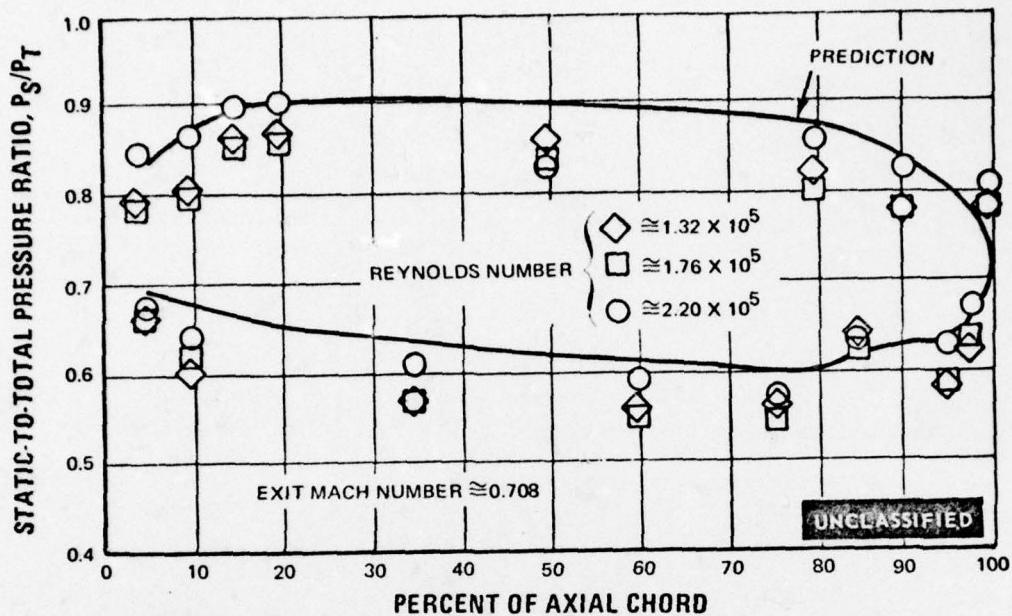


Figure 36 Static-to-Total Pressure Ratio Versus Percent of Axial Chord, Second Blade Root Section; Exit Mach Number = .708

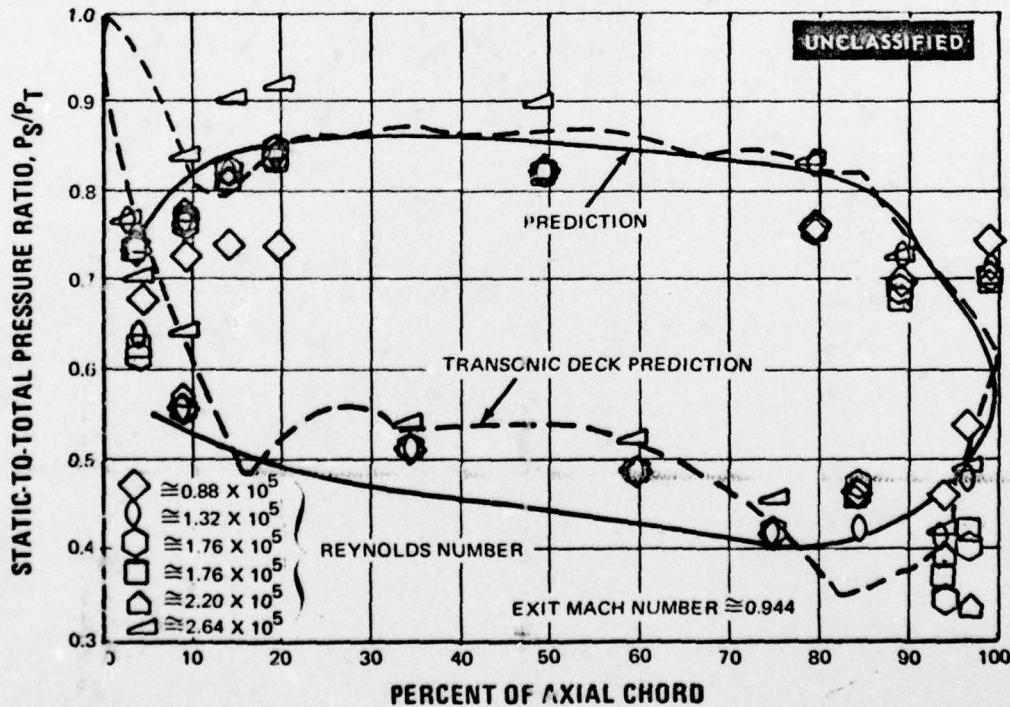


Figure 37 Static-to-Total Pressure Ratio Versus Percent of Axial Chord, Second Blade Root Section; Exit Mach Number = .944

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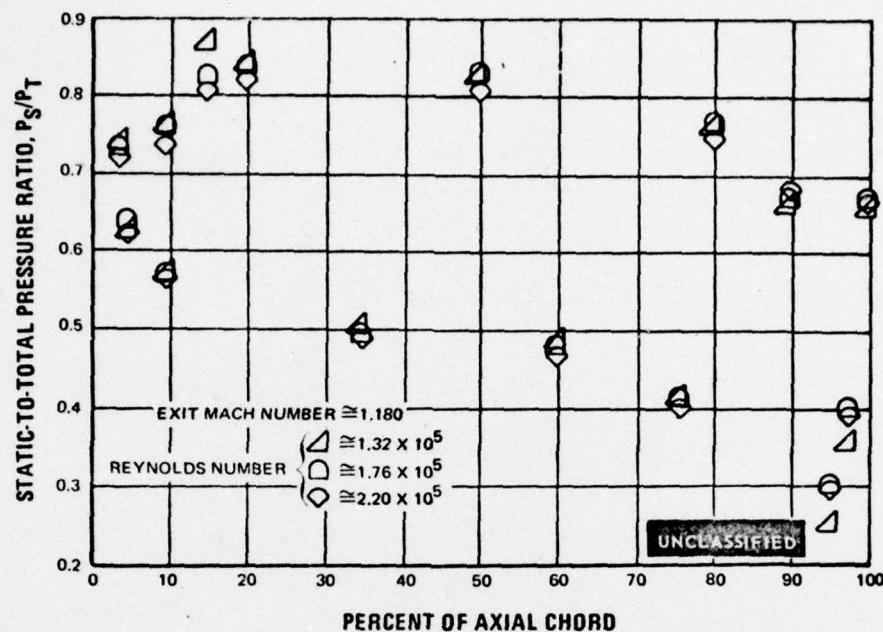


Figure 38 Static-to-Total Pressure Ratio Versus Percent of Axial Chord, Second Blade Root Section; Exit Mach Number = 1.180

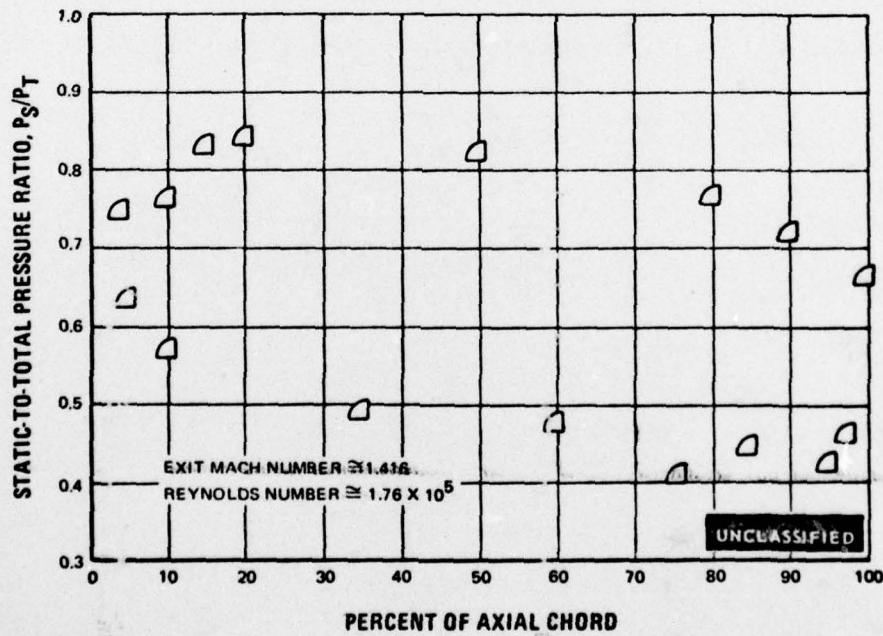


Figure 39 Static-to-Total Pressure Ratio Versus Percent of Axial Chord, Second Blade Root Section; Exit Mach Number = 1.416

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(U) The second blade root section shows a higher profile loss than the combined profile and secondary loss in the blade root section of the part annular cascade test, but interaction between the complex mechanisms occurring in this area are sufficiently complicated that no firm conclusion may be drawn with respect to the result.

## 5. SUMMARY

(U) The midspan section loss coefficients for the first vane, first blade, second vane and second blade measured in the plane cascade were consistent with the mean values measured in the annular segment cascade.

(U) The airfoil root sections which will be designed for the demonstrator two-stage turbine will not have the shock problems exhibited by the tested preliminary airfoil root sections. The new Transonic Airfoil Pressure Distribution Program will be used to design the demonstrator turbine airfoils, which proved to be more accurate for predicting pressure distributions on transonic airfoils.

(U) The objectives of this task were met, since these test results verified the validity of the aerodynamic design procedures that were used to predict the airfoil surface pressure distributions and airfoil boundary layer characteristics.

## 6. TEST PROCEDURE

(U) The plane cascade tests of the medium-reaction normal-solidity airfoils were performed in the Willgoos Laboratory at East Hartford on X-213 stand. The six sections that were tested were the mean sections of the first vane, first blade, second vane, and second blade and the root sections of the second vane and second blade. These performance tests required the use of various combinations of compressors and exhausters, and utilized the heat of compression of the compressors to obtain various combinations of Mach numbers and Reynolds numbers at the cascade exit. Test conditions were always chosen so as to prevent the condensation of atmospheric water vapor.

(U) For each of the six plane sections a thirteen point test program was run. The test program was made to center about the design values of Mach number and Reynolds number, which are listed in Table V. The thirteen point test program is graphically portrayed in Figure 40. At each point the inlet wall, exit wall, and airfoil surface static pressure taps were read and a banjo-type probe was used to survey the exit plane for air angle and total pressure. The procedure listed below was used for these tests:

- The cascade pack was installed on the stand and all of the fixed instrumentation was connected
- The traversing probe was installed at the exit plane of the cascade pack and was correctly located with respect to the instrumented airfoil
- The total pressure transducers for the traversing probe and the data recording equipment were calibrated

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- The laboratory compressors and exhausters which were required for the test were started and allowed to run until steady operating conditions were reached, which took about two hours
- Preceeding the test, the valves upstream of the cascade rig were closed and the exhausters were used to reduce the exit pressure to an arbitrary low value; in this manner any faulty static pressure taps were discovered
- In turn, each of the thirteen test points were set and a complete set of data was recorded

TABLE V

## DESIGN MACH NUMBERS AND REYNOLDS NUMBERS FOR THE SIX PLANE CASCADE SECTIONS

Section	Mach Number	Reynolds number
First vane mean	0.852	$4.38 \times 10^5$
First blade mean	0.788	$2.26 \times 10^5$
Second vane root	1.003	$2.86 \times 10^5$
Second vane mean	0.870	$2.85 \times 10^5$
Second blade root	0.944	$1.76 \times 10^5$
Second blade mean	0.904	UNCLASSIFIED
		$1.52 \times 10^5$

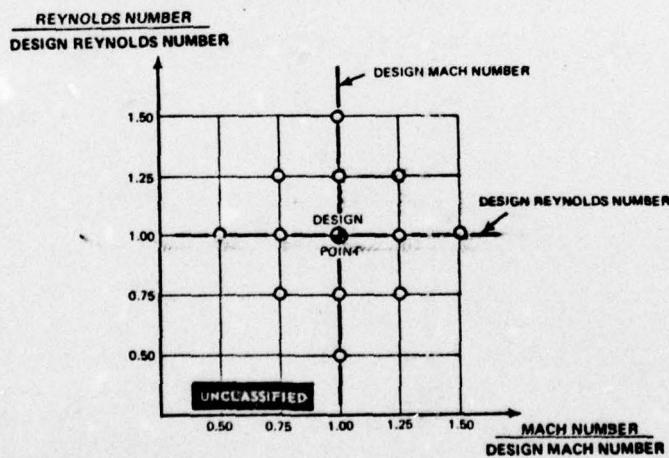


Figure 40 Plane Cascade Test Program

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## SECTION IV

### MEDIUM SOLIDITY AIRFOIL EVALUATION (TASK II<sup>d</sup>)

#### 1. OBJECTIVE

(U) The initial objective of Task II<sup>d</sup> was to investigate two additional boundary layer control methods, other than the two methods that were to be investigated under Task II<sup>c</sup>, on two different airfoils. Since the performance of the four baseline airfoils was very similar (Reference 3), Task II<sup>c</sup> was modified so that four boundary layer control methods were applied only to the second vane (Reference 4, Section IV), eliminating the need for Task II<sup>d</sup> as originally conceived. By mutual agreement between Pratt & Whitney Aircraft and the Air Force, the work substituted for the original Task II<sup>d</sup> was the evaluation of the performance of lower-solidity/higher-load-coefficient airfoils designed for the same velocity triangles as the baseline airfoils of Task II<sup>b</sup>. The performance of the first vane and first blade lower-solidity airfoils was reported in Reference 4, and the second vane and second blade results, as well as a summary for all four airfoils investigated, are presented in this Section.

#### 2. TASK OBJECTIVE

(U) Each of the four medium solidity airfoils were evaluated in an annular segment cascade exactly as in Task II<sup>b</sup> (Reference 3). The data has been analyzed and the task objectives were met by the following steps:

- Measurement of all important aerodynamic properties at the cascade inlet and exit planes
- Reconstruction of the entire exit plane loss distribution
- Reconstruction of the entire exit plane flow pattern
- Measurement of the airfoil surface static pressure distributions at three radial locations
- Careful analysis of all data and visual clues

#### 3. AIRFOIL SECTION AND FACILITY DESIGN

(U) The medium-reaction, medium-solidity airfoils were designed to the same turbine velocity diagrams as the normal solidity airfoils. These diagrams were presented in Reference 1. A summary of the pertinent design values, the airfoil elevations, gaging distribution, airfoil sections, predicted surface pressure distributions, and airfoil radius of curvature distributions for each of the four medium-solidity airfoils was presented at five spanwise locations in Reference 3. The four airfoils are the first and second vanes, and the first and second blades. The fabrication coordinates of each airfoil were also tabulated in Reference 3, including the airfoil angles, airfoil areas, axial chords and uncovered turnings.

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(U) The details of the test section design, location of static pressure taps and the method of installation, details of the traversing probes and their calibration, location of the traversing planes, and probe corrections for pitch angle were presented in Reference 4. The axial location dimensions for the traversing planes and end wall static pressure taps were omitted in Reference 4. The following list gives the locations for planes M-M, N-N, O-O, and P-P shown in Reference 4, Figures 142 and 144.

- Plane M-M is located 1.00 inch axially upstream of the test vane tip section or test blade root section leading edge
- Plane N-N is located 0.788 inch axially downstream from plane M-M
- Plane O-O is located 0.185 inch axially downstream of the test airfoil trailing edge at the mean section C-C
- Plane P-P is located 0.625 inch axially downstream of the test airfoil trailing edge at the mean section C-C

## **4. DISCUSSION**

(U) The partial results of this task have been reported in Reference 4, since at the time of publication of that report the performance of the medium solidity second vane and second blade had not been evaluated. This work has now been completed and will be reported herein. In addition, a comparison of the performance of the nominal and medium solidities will also be made.

(U) As previously reported (Reference 4) the inlet guide vanes for the second vane and second blade did not produce the correct inlet angle to their respective test airfoils. This deficiency was substantially improved by a reworking of these guide vanes and the new results are shown in Figures 41 and 42. The associated inlet total pressure spanwise pressure loss distributions are shown in Figures 43 and 44. The inlet total pressure loss contours from which these were obtained are shown in Figures 45 and 46.

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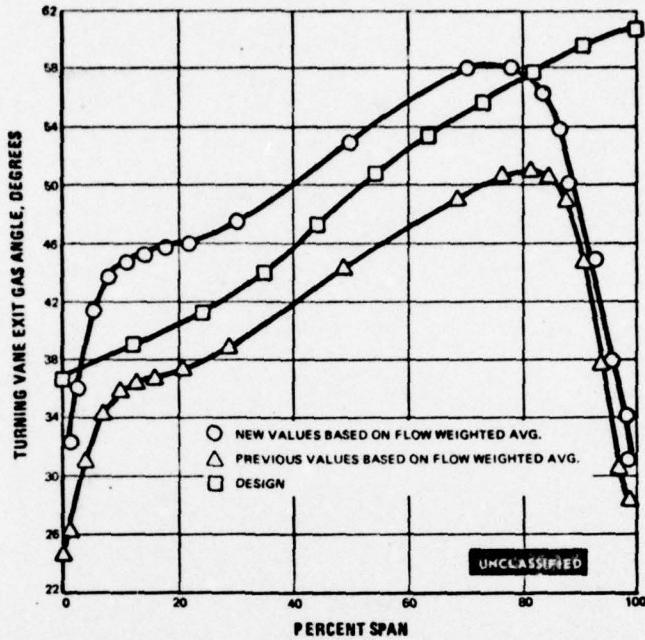


Figure 41 Inlet Guide Vane Spanwise Exit Angle Distribution, Second Vane, Medium Solidity Cascade - Midspan Exit Test Airfoil Mach No. = 0.820

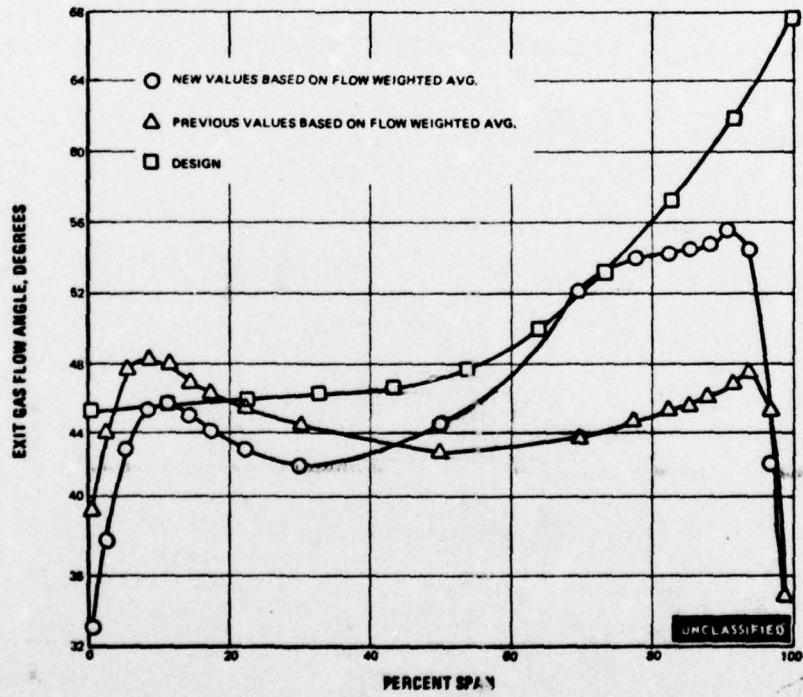
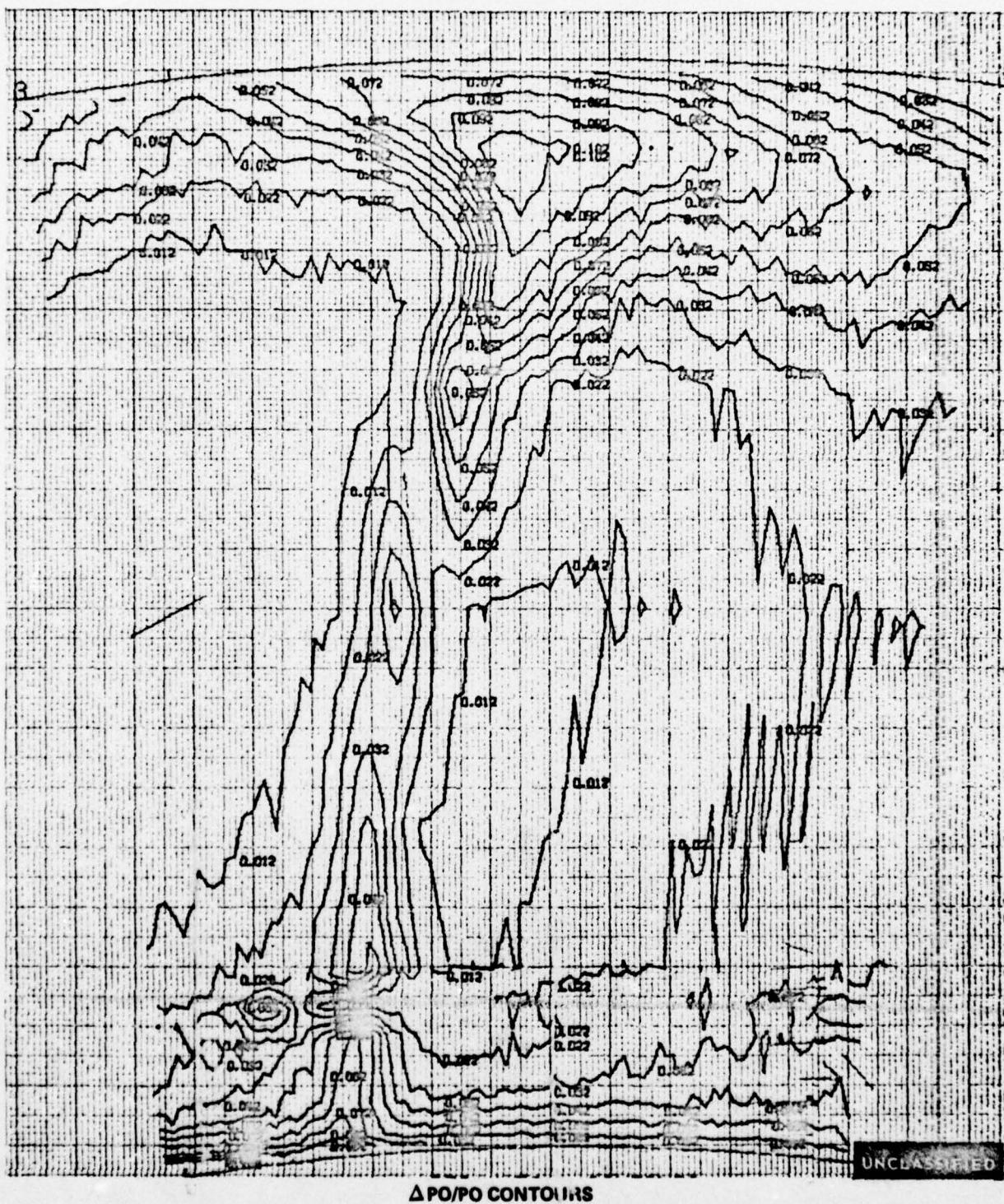


Figure 42 Inlet Guide Vane Spanwise Exit Angle Distribution, Second Blade, Medium Solidity Cascade - Midspan Exit Test Airfoil Mach No. = 0.940

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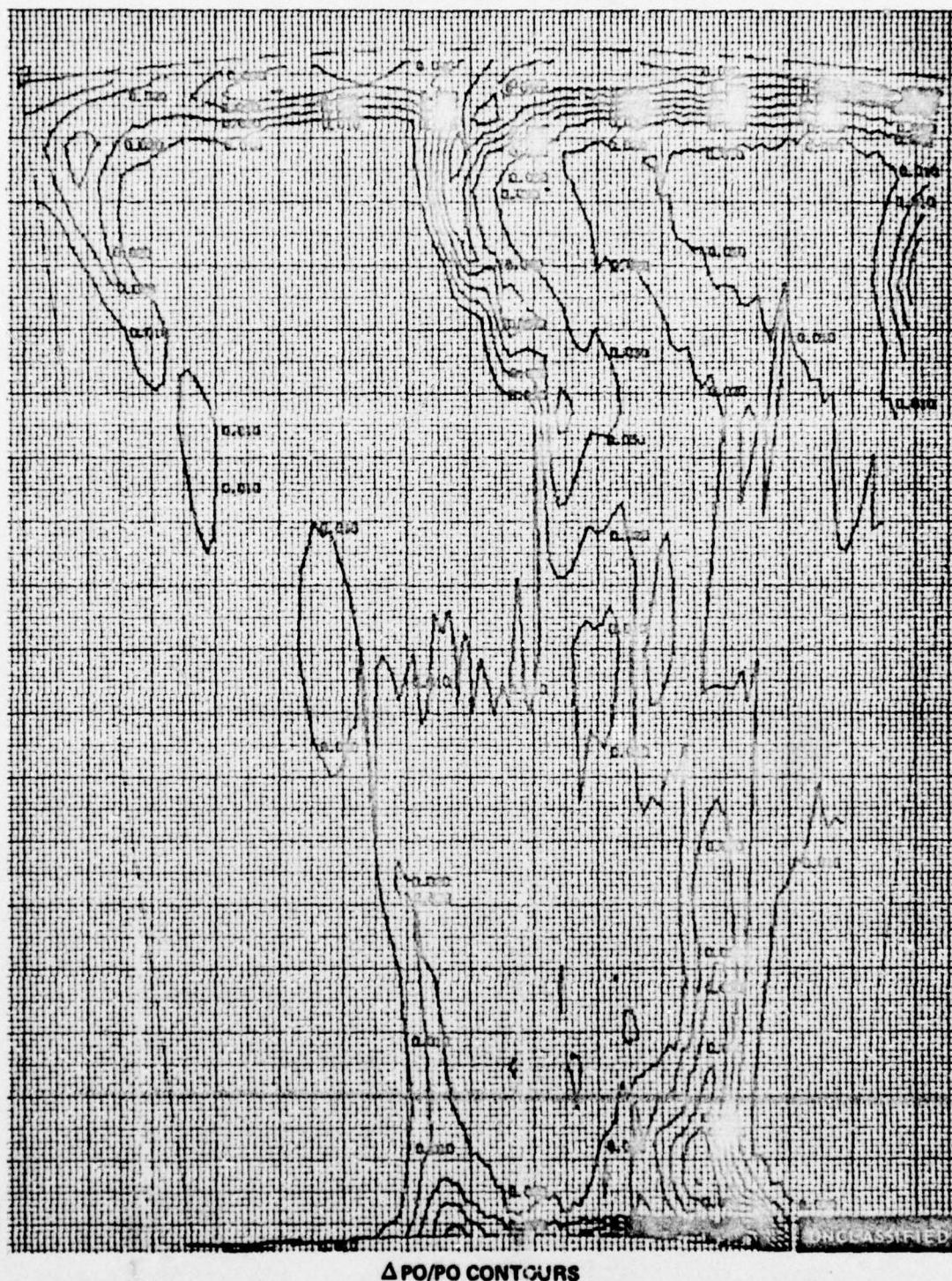
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**Figure 43** Inlet Guide Vane Pressure Loss Contours, Second Vane, Medium Solidity Cascade - Midspan Exit Airfoil Mach No. = 0.820

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**Figure 44      Inlet Guide Vane Pressure Loss Contours, Second Blade, Medium Solidity Cascade, Midspan Exit Test Airfoil Mach No. = 0.940**

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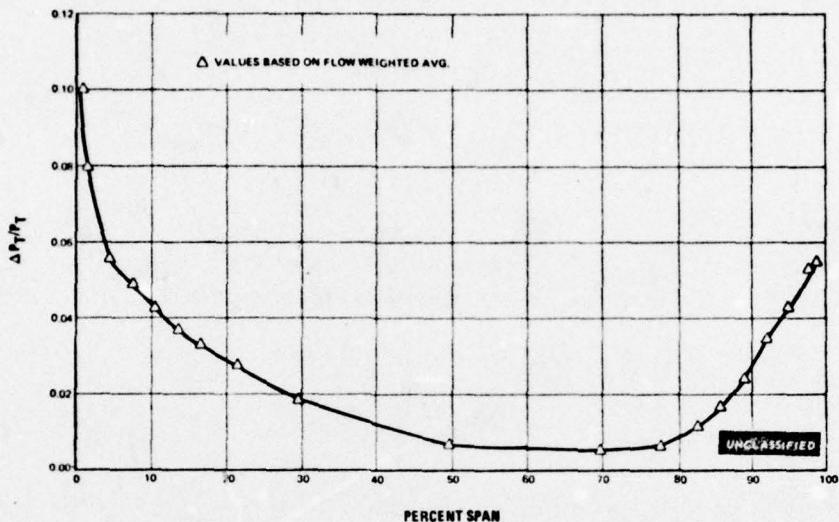


Figure 45 Inlet Guide Vane Spanwise Pressure Loss Distribution, Second Vane, Medium Solidity Cascade, Midspan Exit Test Airfoil Mach No. = 0.820

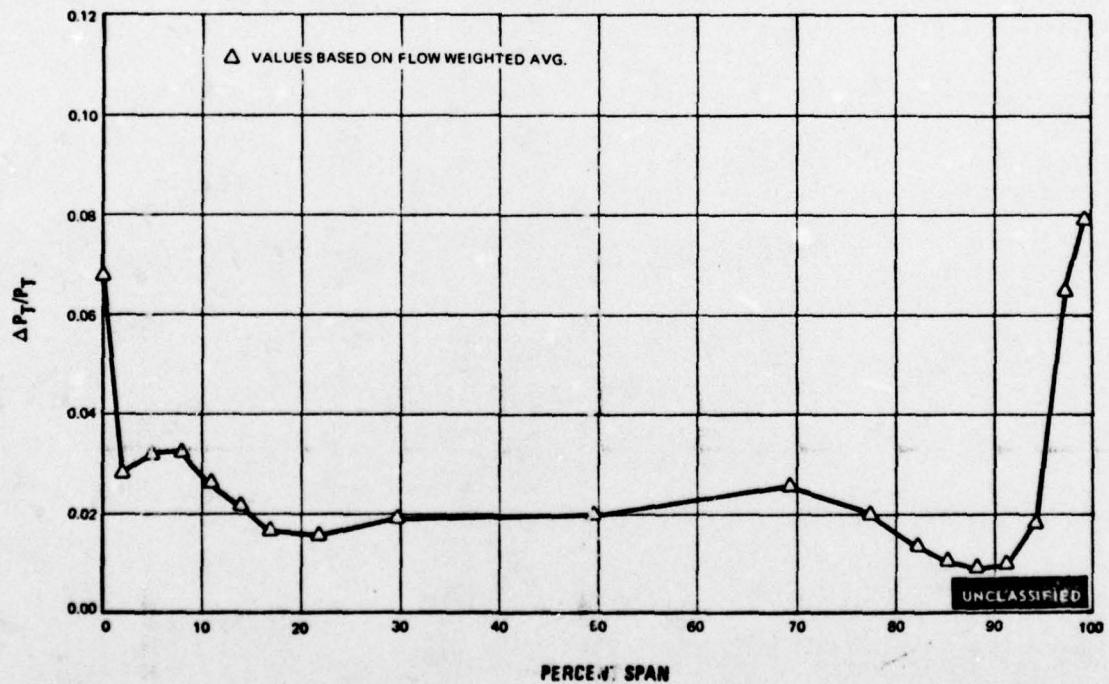


Figure 46 Inlet Guide Vane Spanwise Pressure Loss Distribution, Second Blade, Medium Solidity Cascade, Midspan Exit Test Airfoil Mach No. 0.940

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(U) The second vane data indicates that the airfoils are separated over the majority of the span. The contour plot of loss ( $\frac{\Delta P_o}{P_o}$ ) (Figure 47) shows distinct wakes only near

the root region and the spanwise flow-weighted average values of  $\frac{\Delta P_o}{P_o}$  and  $1-\phi^2$  shown in

Figures 48 and 49 are very high over most of the span. The low  $1-\phi^2$  values shown at the root and tip are partially due to the method of accounting for the spanwise inlet total pressure profile. The low total pressure fluid at the end walls of the inlet plane, assumed to remain at the same percent of the span, is actually convected radially inward to the quarter root and quarter tip regions of the exit plane. Therefore, a more meaningful average loss is found by flow-weighting the loss from the root-to-mean and mean-to-tip sections. These second vane averages are 0.085 and 0.234, respectively. The measured spanwise flow-weighted exit gas angle (Figure 50) and the exit gas angle contours (Figure 51) for the second vane show higher than predicted values particularly toward the tip section and the spanwise flow-weighted exit Mach number distribution shown in Figure 52 is lower than predicted. Both of these effects can be traced to airfoil boundary layer separation.

(U) The measured and predicted airfoil static pressure distributions are shown in Figures 53, 54 and 55 for the second vane root, mean and tip sections. The mean section test values show good agreement with predictions. The pressures in the throat-to-trailing-edge appear to be lower than predicted. This is likely due to the fact that the predictions of the Airfoil Pressure Distribution Program are inaccurate when extrapolated into the transonic flow regime. The tip section also shows good agreement except near the leading edge due to positive incidence at this section. The root is unloaded relative to the prediction due to the static pressure conditions at the exit plane. This situation was discussed in detail in Reference 3.

(U) The second blade data indicates that the airfoils are performing well from the root to the mean section and that the airfoils are separated from the mean to the tip section. The contour plot of  $\Delta P_o/P_o$  in Figure 56, and the spanwise loss distribution in Figure 57 clearly show distinct wakes and low losses from the root to the mean section. The spanwise flow weighted values of  $1-\phi^2$  shown in Figure 58 are also low from the root to the mean and increasingly higher from the mean to the tip section. The spanwise Mach number distributions (Figure 59) and calculated values of spanwise flow coefficient indicated that separation occurred in the tip region of this airfoil. The plots of exit gas angle contours (Figure 60) and spanwise flow-weighted exit gas angle (Figure 61) did not support the possibility of complete separation, however; implying perhaps that reattachment took place.

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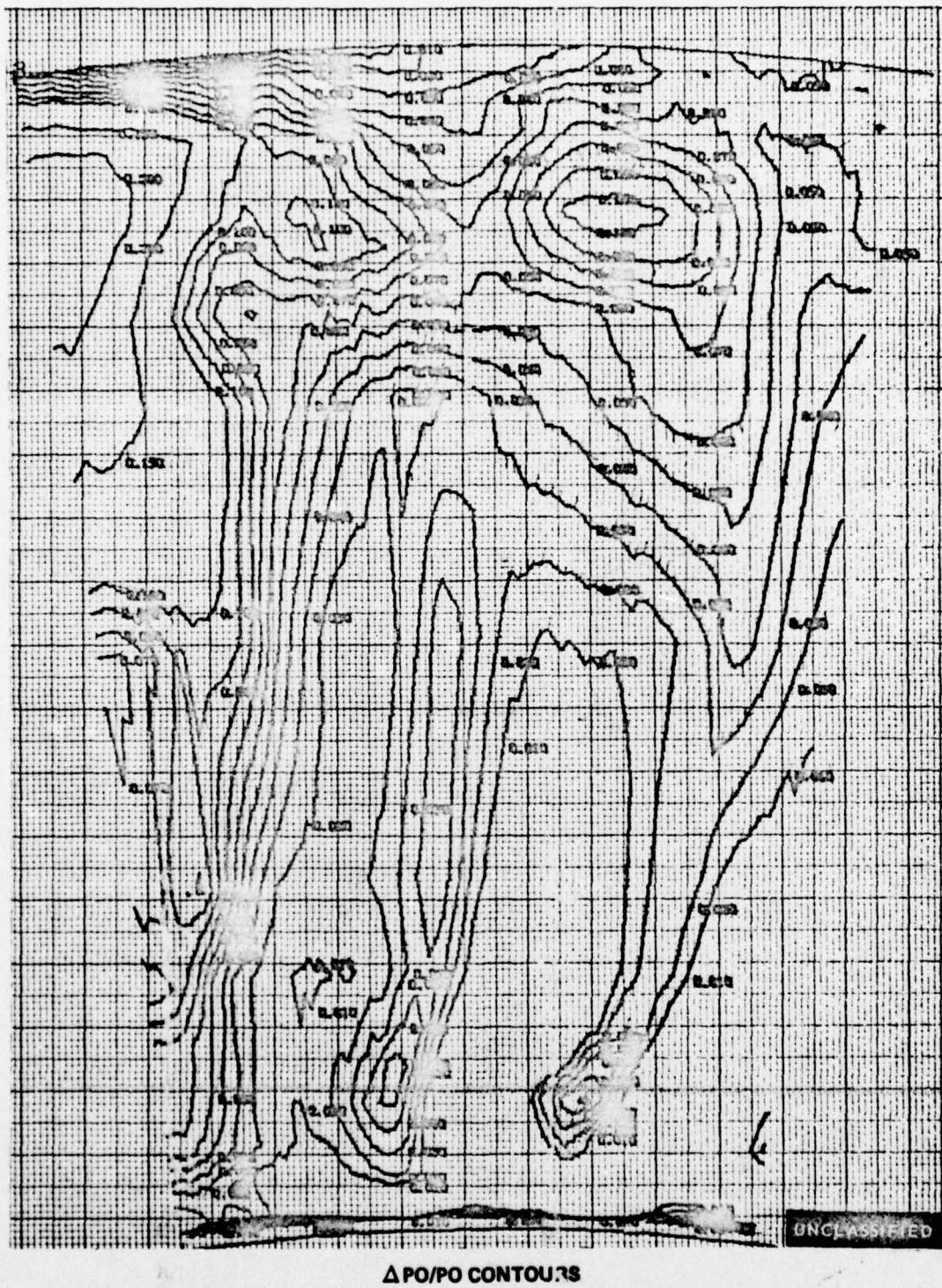


Figure 47 Pressure Loss Contours, Second Vane, Medium Solidity, Three Flow Passages, Midspan Exit Mach No. = 0.820

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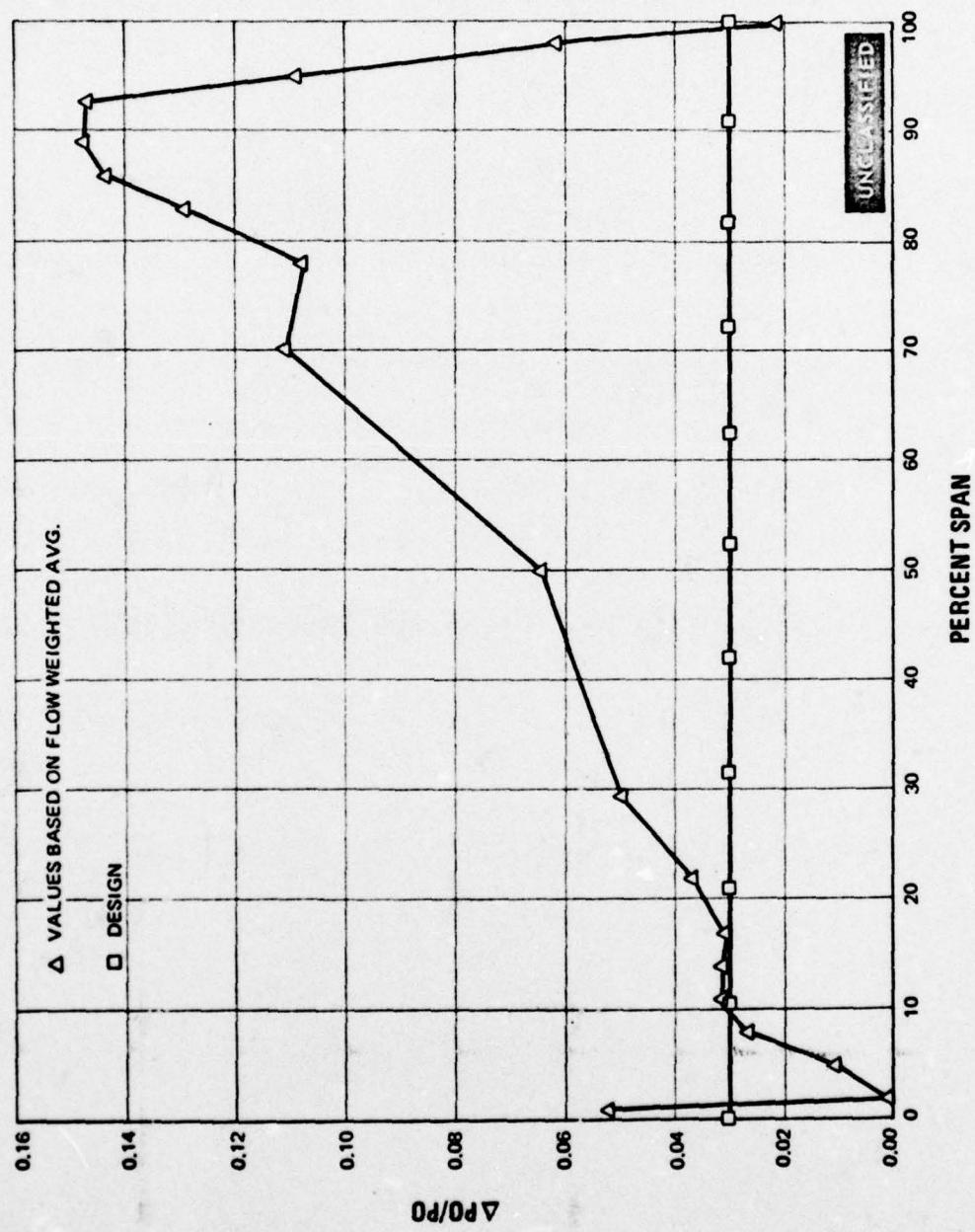


Figure 48 Spanwise Pressure Loss Distribution, Second Vane, Medium Solidity, Midspan Exit Mach No. = 0.820

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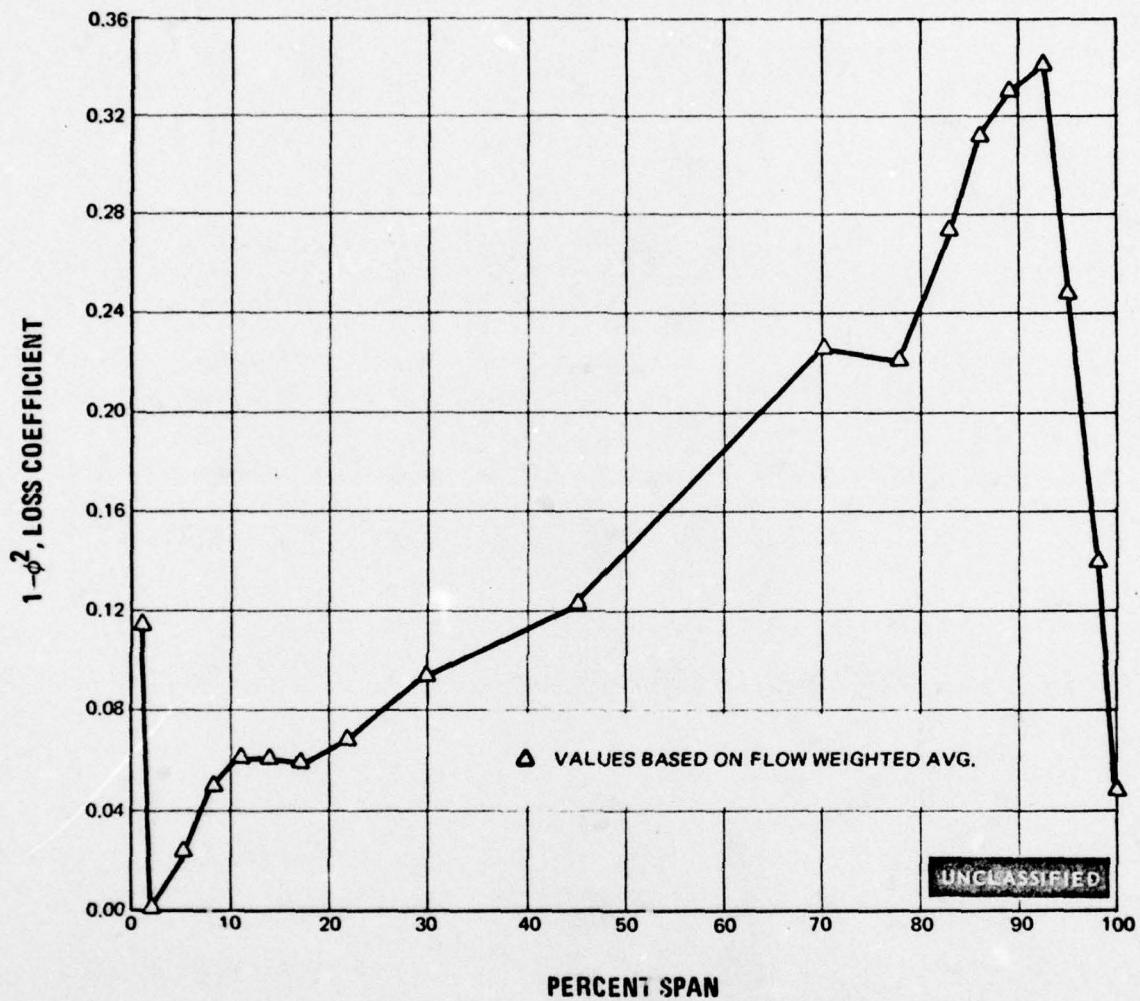


Figure 49 Spanwise Loss Coefficient Distribution, Second Vane, Medium Solidity, Mid-span Exit Mach No. = 0.820

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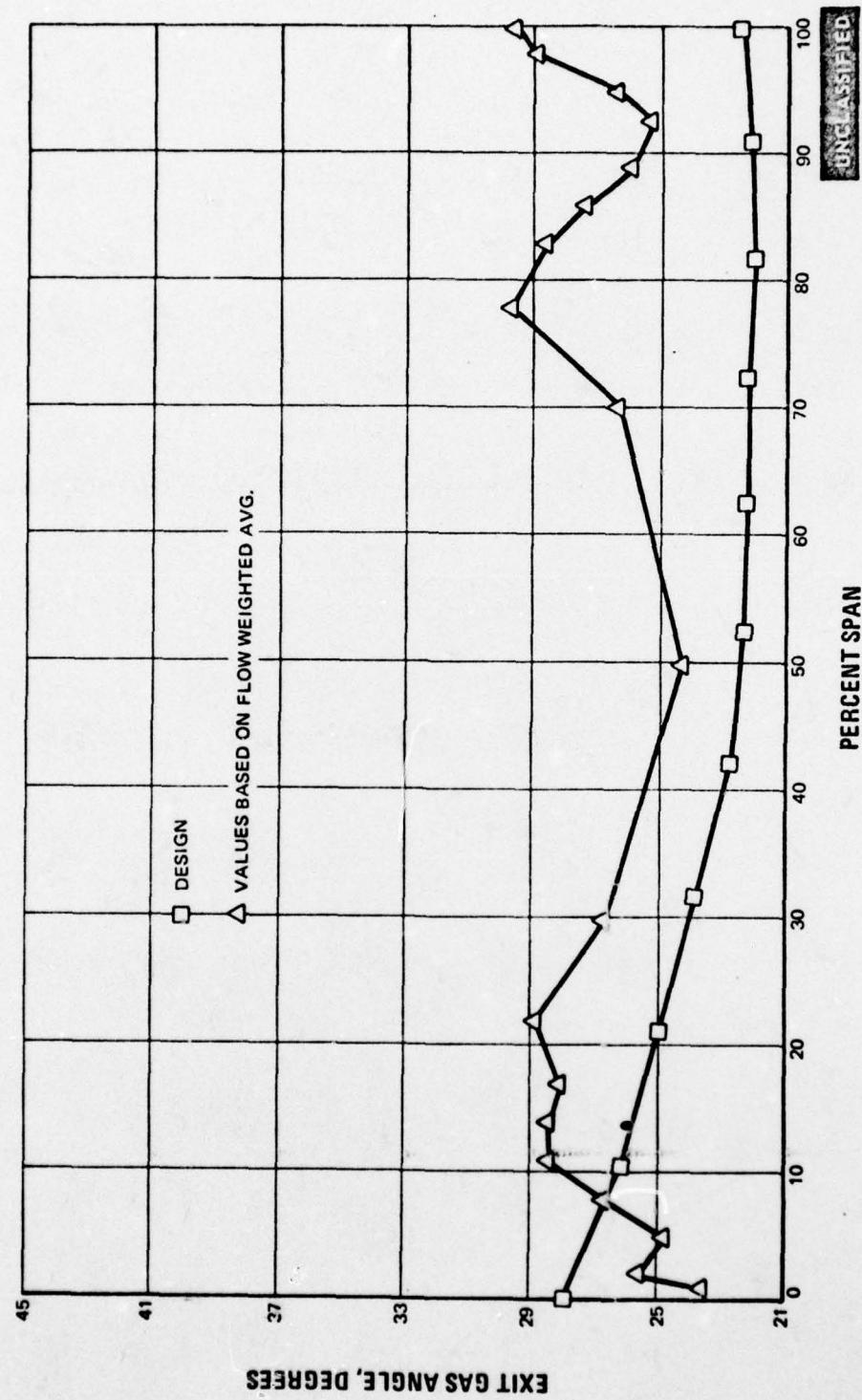
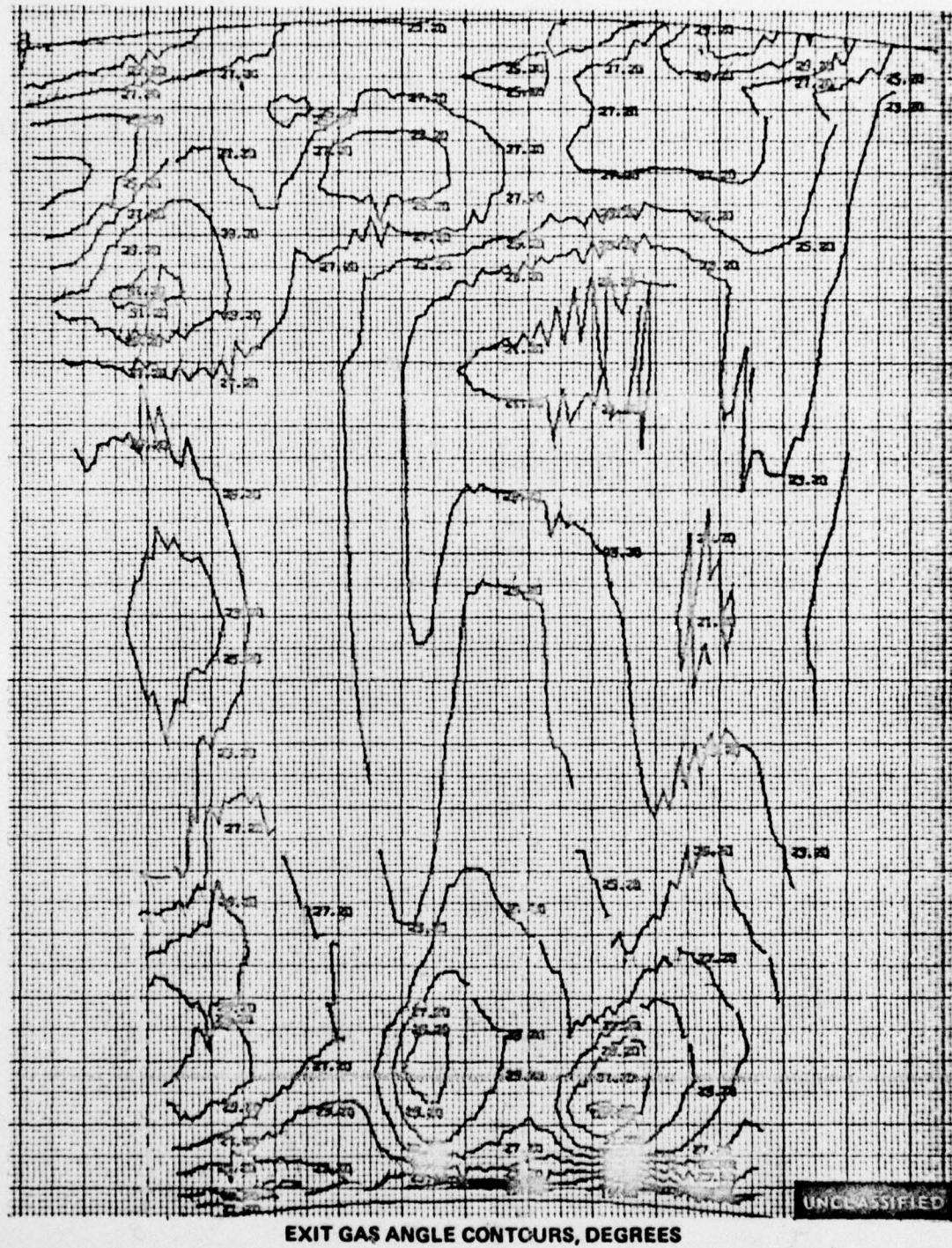


Figure 50 Spanwise Exit Gas Angle Distribution, Second Vane, Medium Solidity, Mid-span Exit Mach No. = 0.820

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**Figure 51**    Exit Gas Angle Contours, Second Vane, Medium Solidity, Three Flow Passages, Midspan Exit Mach No. = 0.820

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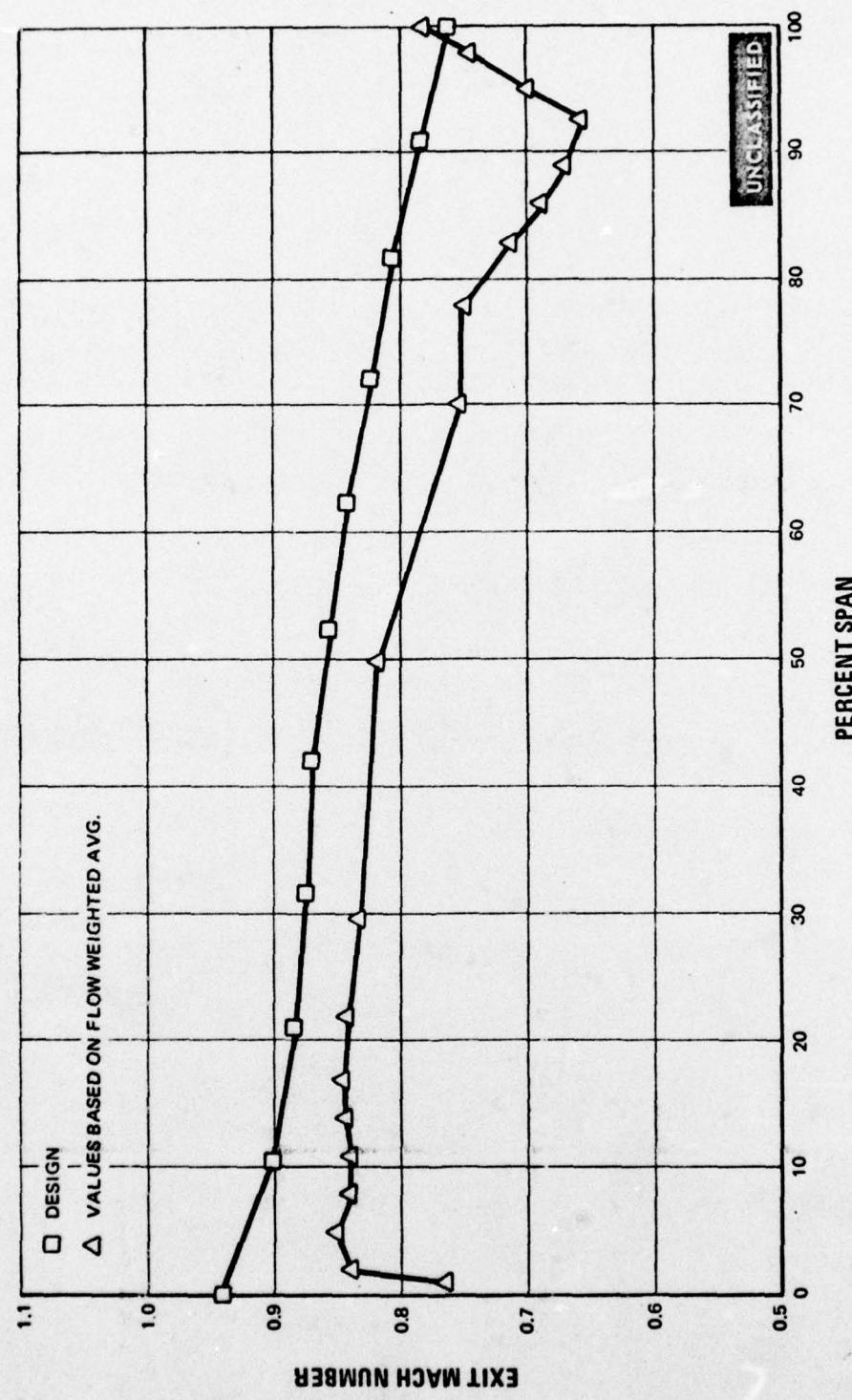


Figure 52      Spanwise Exit Mach Number Distribution, Second Vane, Medium Solidity,  
Midspan Exit Mach No. = 0.820

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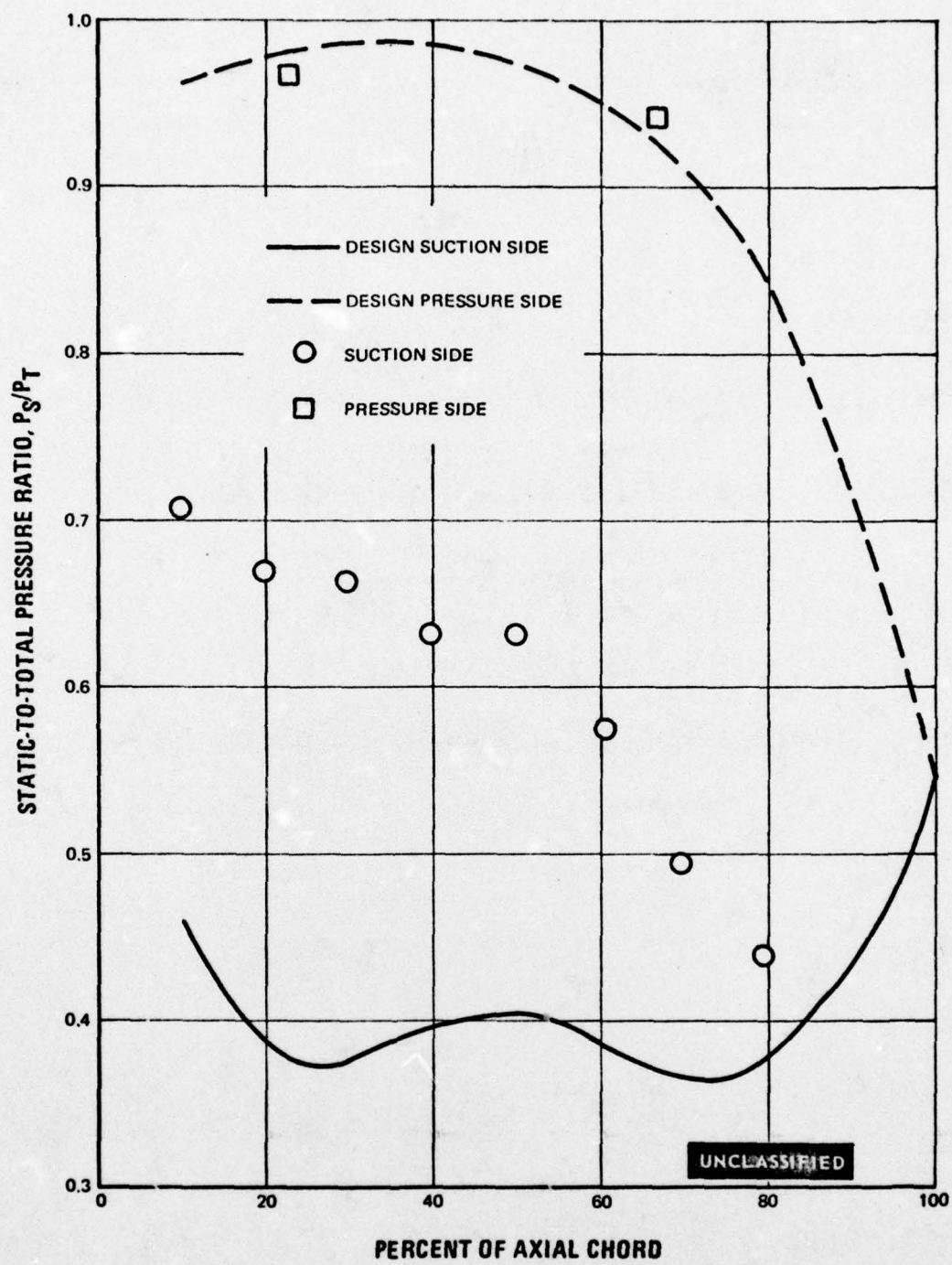
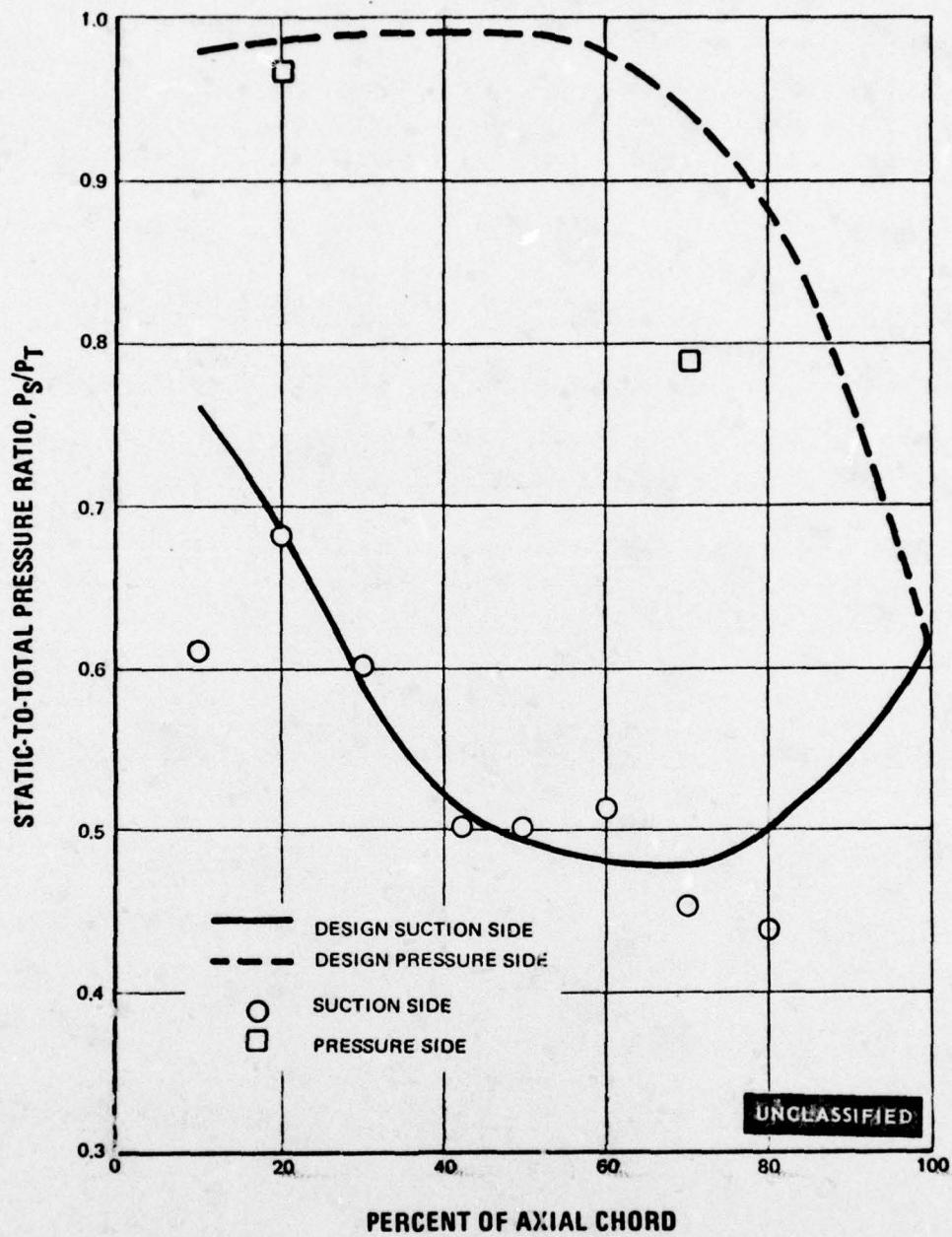


Figure 53      Static-to-Total Pressure Ratio Versus Percent Axial Chord, Second Vane,  
Medium Solidity, Root Section

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**Figure 54**      Static-to-Total Pressure Ratio Versus Percent Axial Chord, Second Vane,  
Medium Solidity, Mean Section

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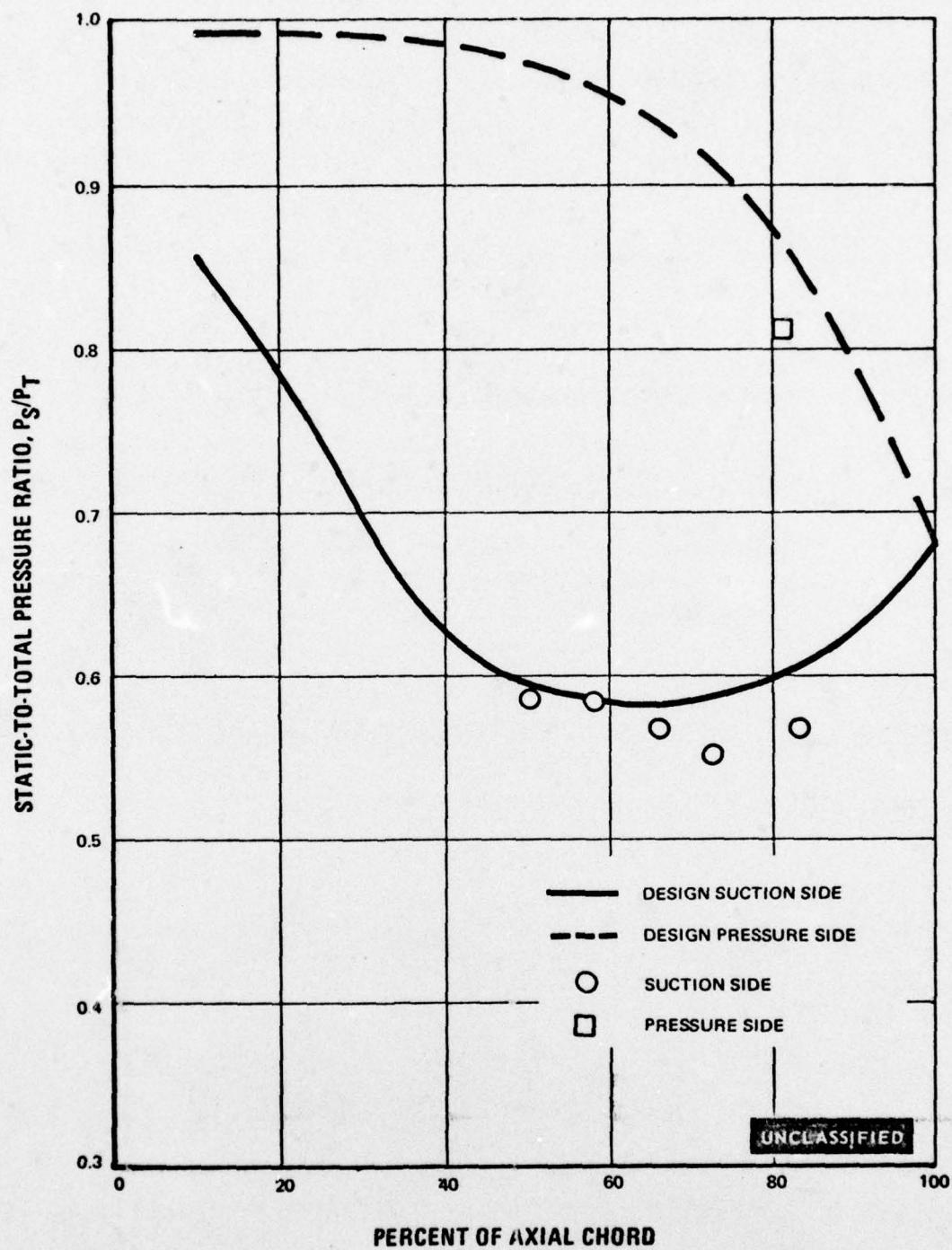
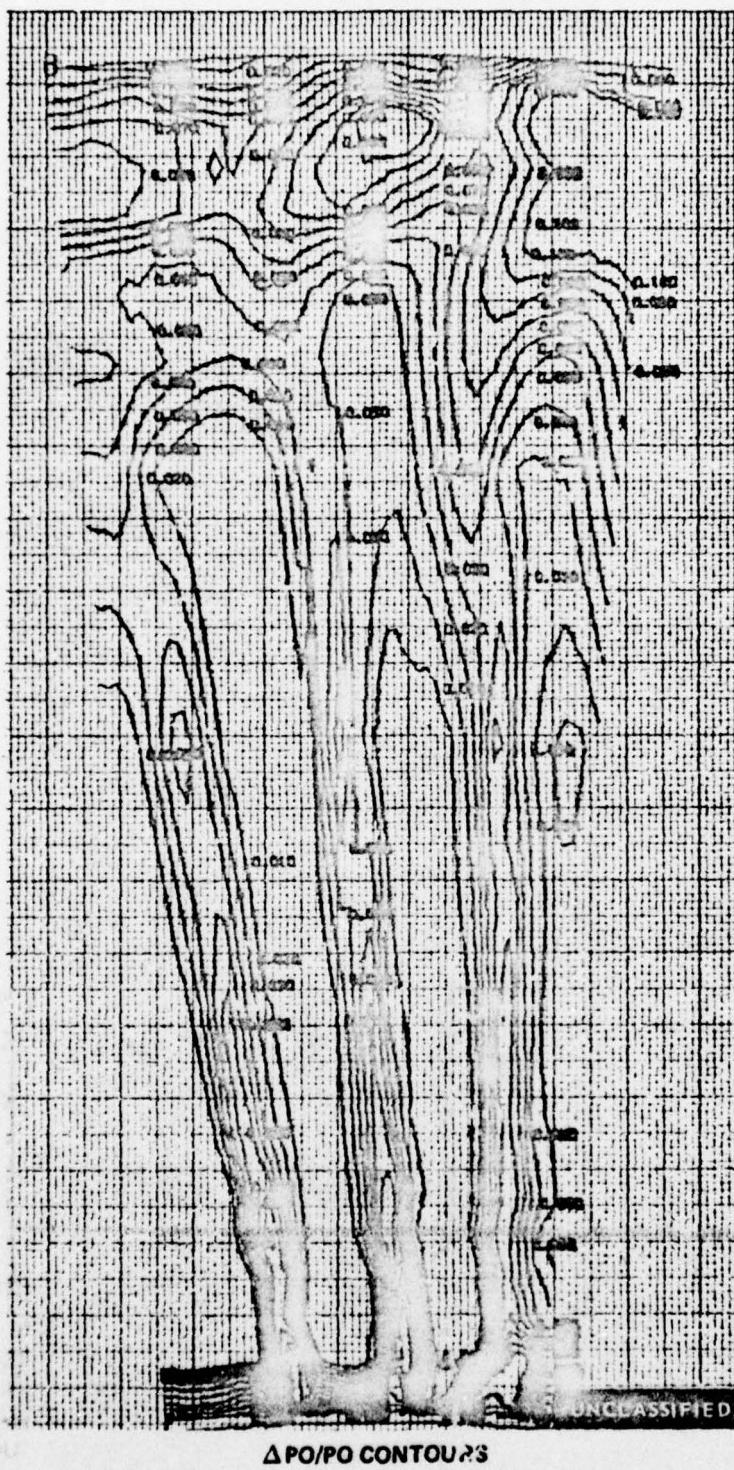


Figure 55      Static-to-Total Pressure Ratio Versus Percent Axial Chord, Second Vane,  
Medium Solidity, Tip Section

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**Figure 56 Pressure Loss Contours, Second Blade Medium Solidity, Three Flow Passages,  
Midspan Exit Mach No. = 0.940**

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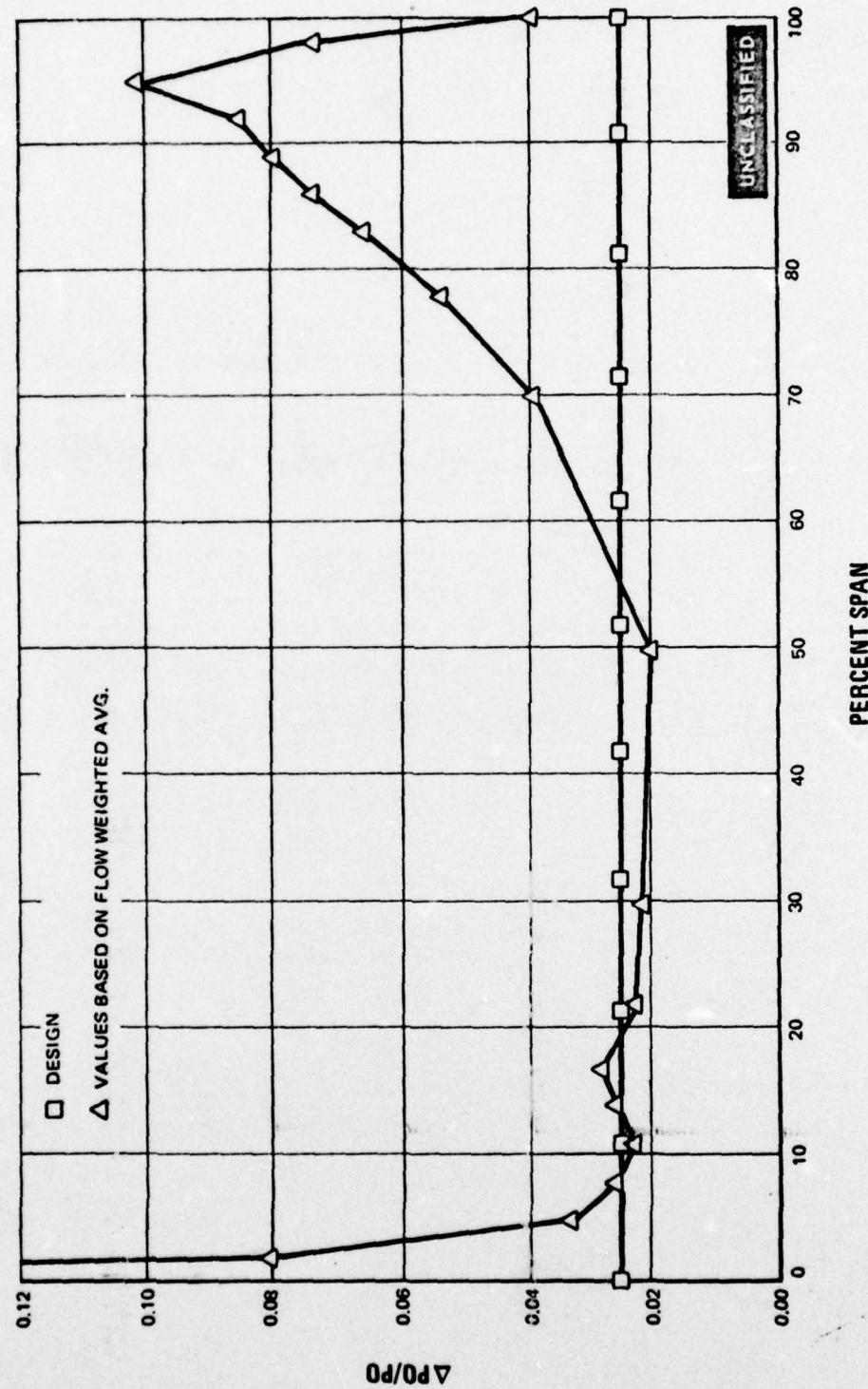


Figure 57 Spanwise Pressure Loss Distribution, Second Blade, Medium Solidity, Midspan Exit  
Mach No. = 0.940

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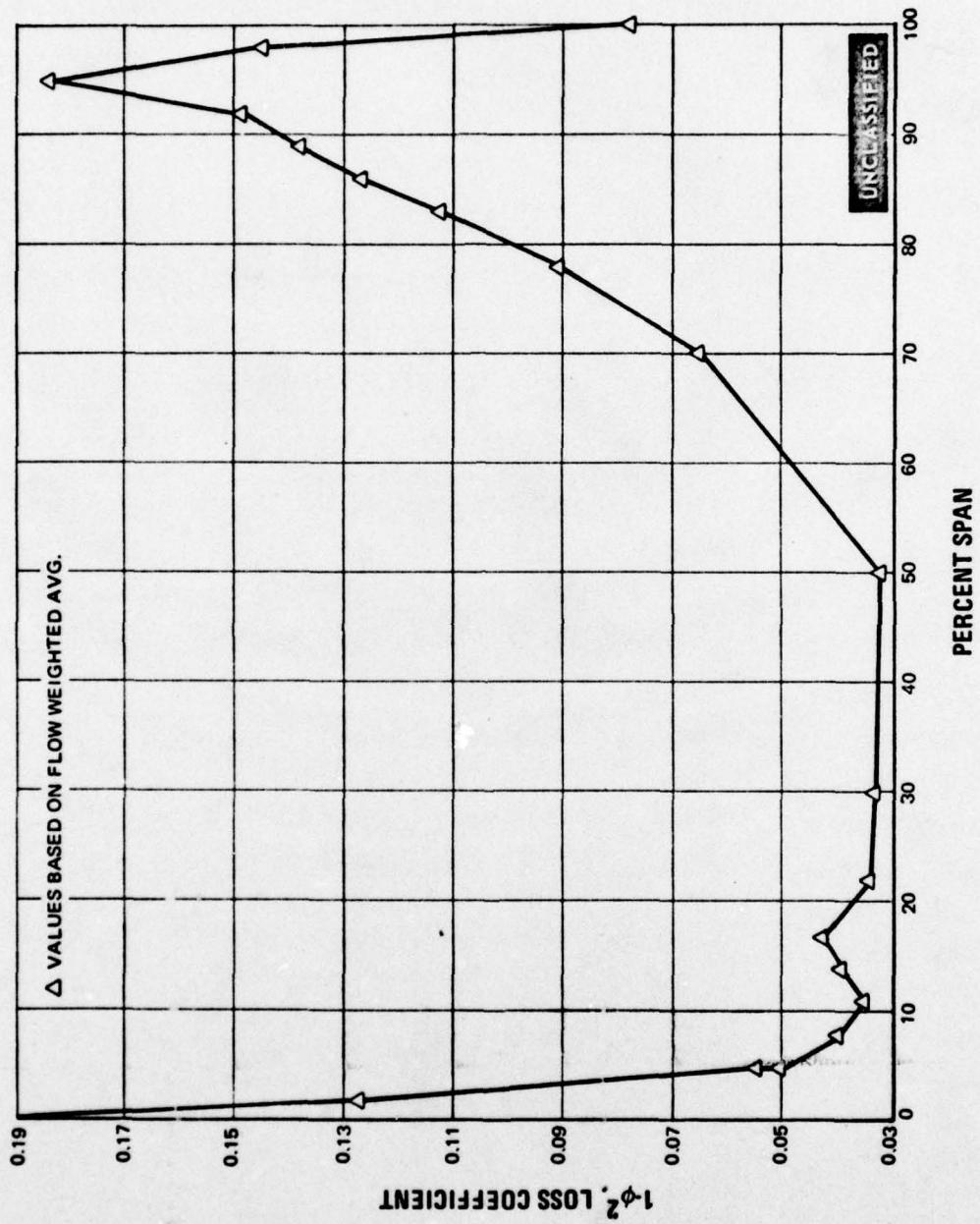


Figure 58 Spanwise Loss Coefficient Distribution, Second Blade, Medium Solidity,  
Midspan Exit Mach No. = 0.940

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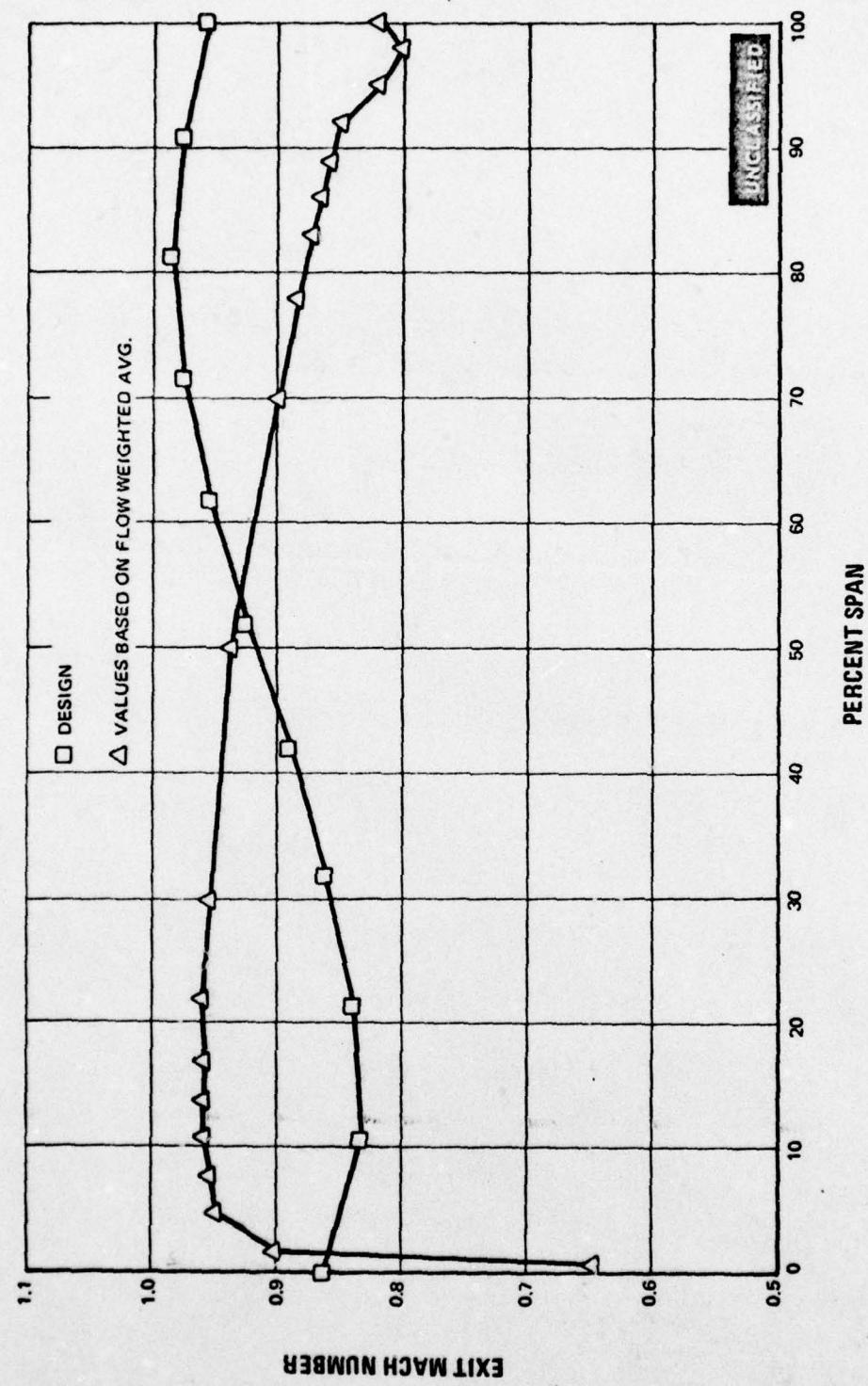
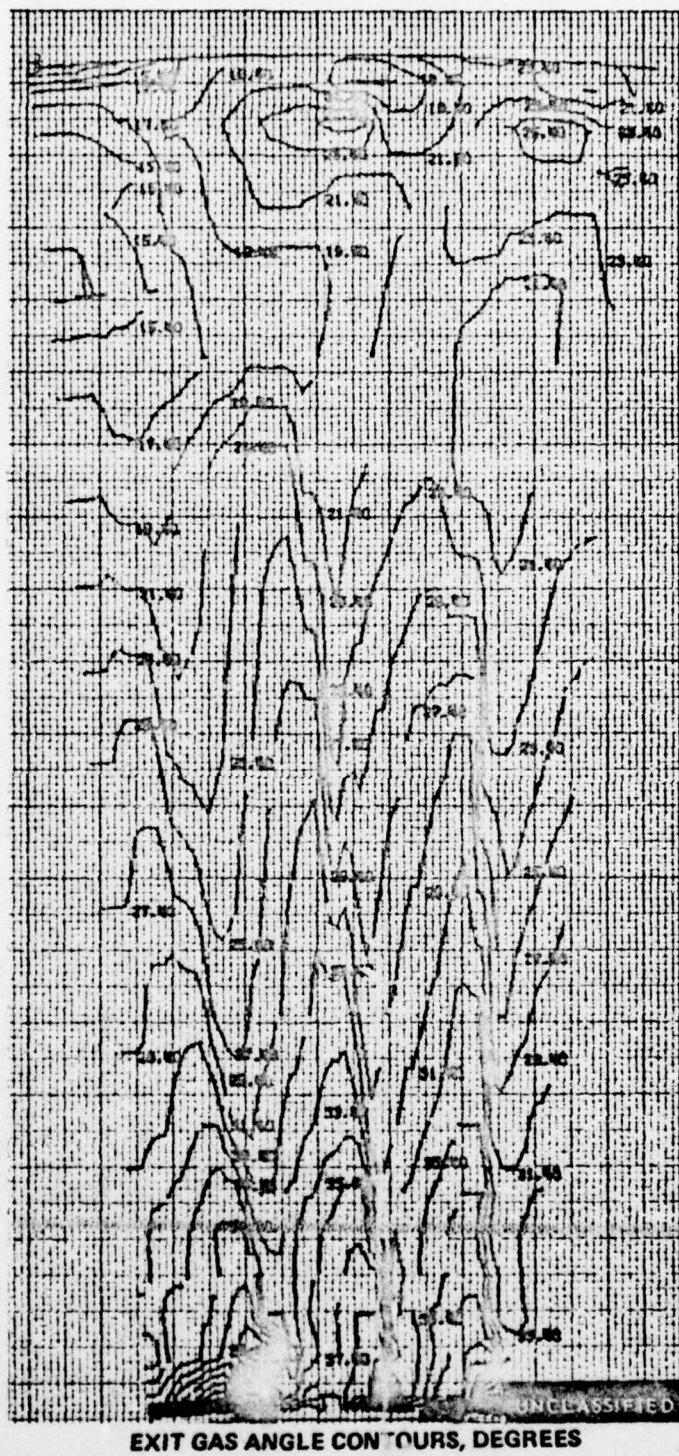


Figure 59 Spanwise Exit Mach Number Distribution, Second Blade, Medium Solidity,  
Midspan Exit Mach No. = 0.940

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**Figure 60**      Exit Gas Angle Contours, Second Blade, Medium Solidity, Three Flow Passages, Midspan Exit Mach No. = 0.940

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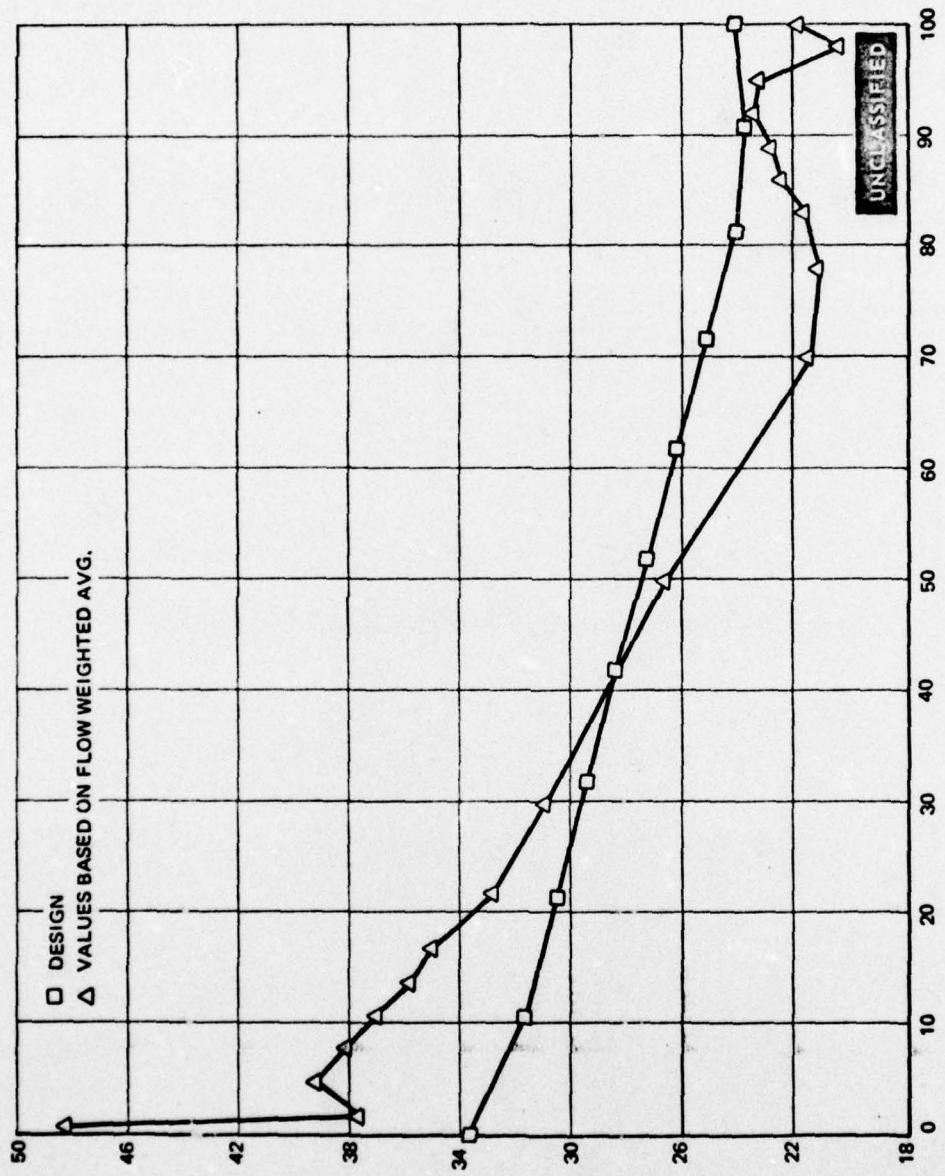


Figure 61 Spanwise Exit Gas Angle Distribution, Second Blade, Medium Solidity, Mid-span Exit Mach No. = 0.940

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(U) The measured and predicted airfoil pressure distributions are shown in Figures 62, 63 and 64. From the data available it may be concluded that the mean section was performing as predicted. Although one data point agrees with prediction, the inlet angle profile gives evidence that the tip section was operating at positive incidence of at least 10 degrees and, therefore, the actual pressure distribution would show lower suction side values of  $P_s/P_t$  than predicted near the leading edge. The root section is unloaded relative to the prediction for the usual reasons.

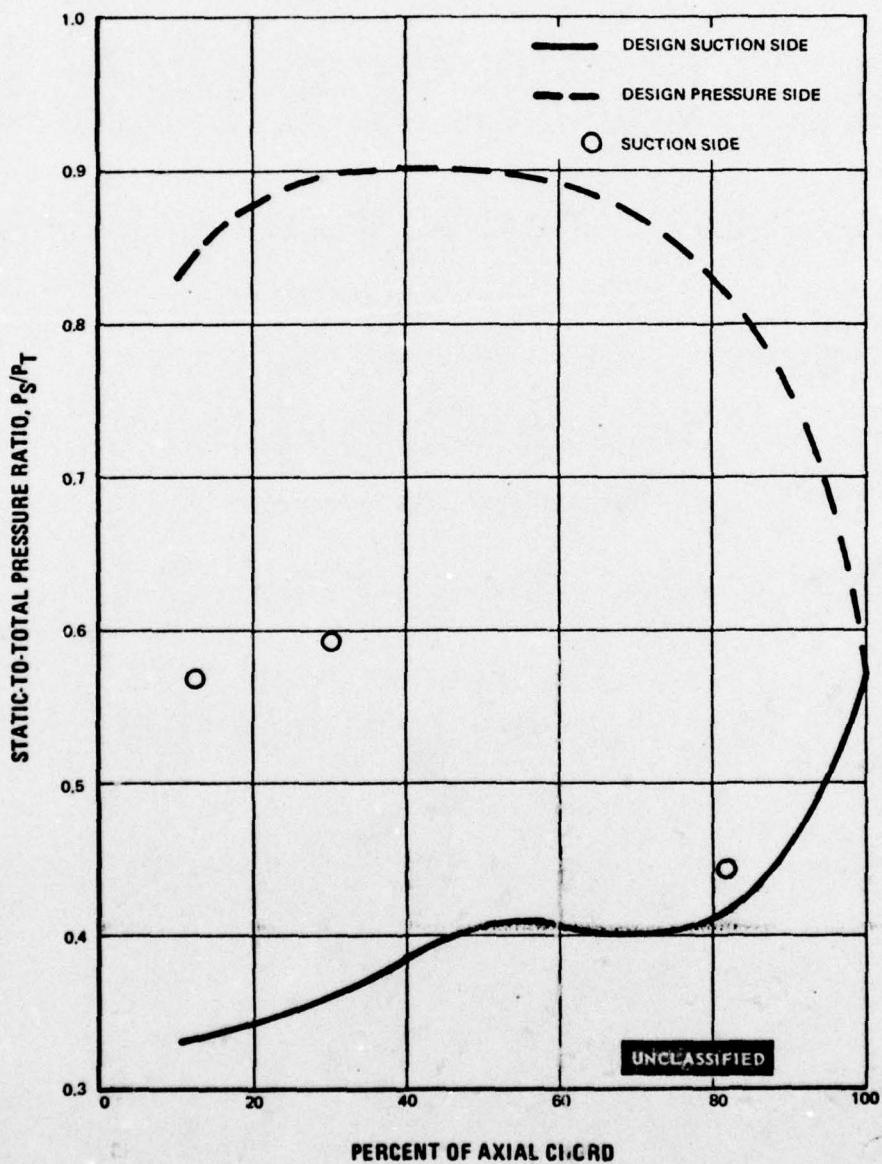


Figure 62      Static-to-Total Pressure Ratio Versus Percent Axial Chord, Second Blade, Medium Solidity, Root Section

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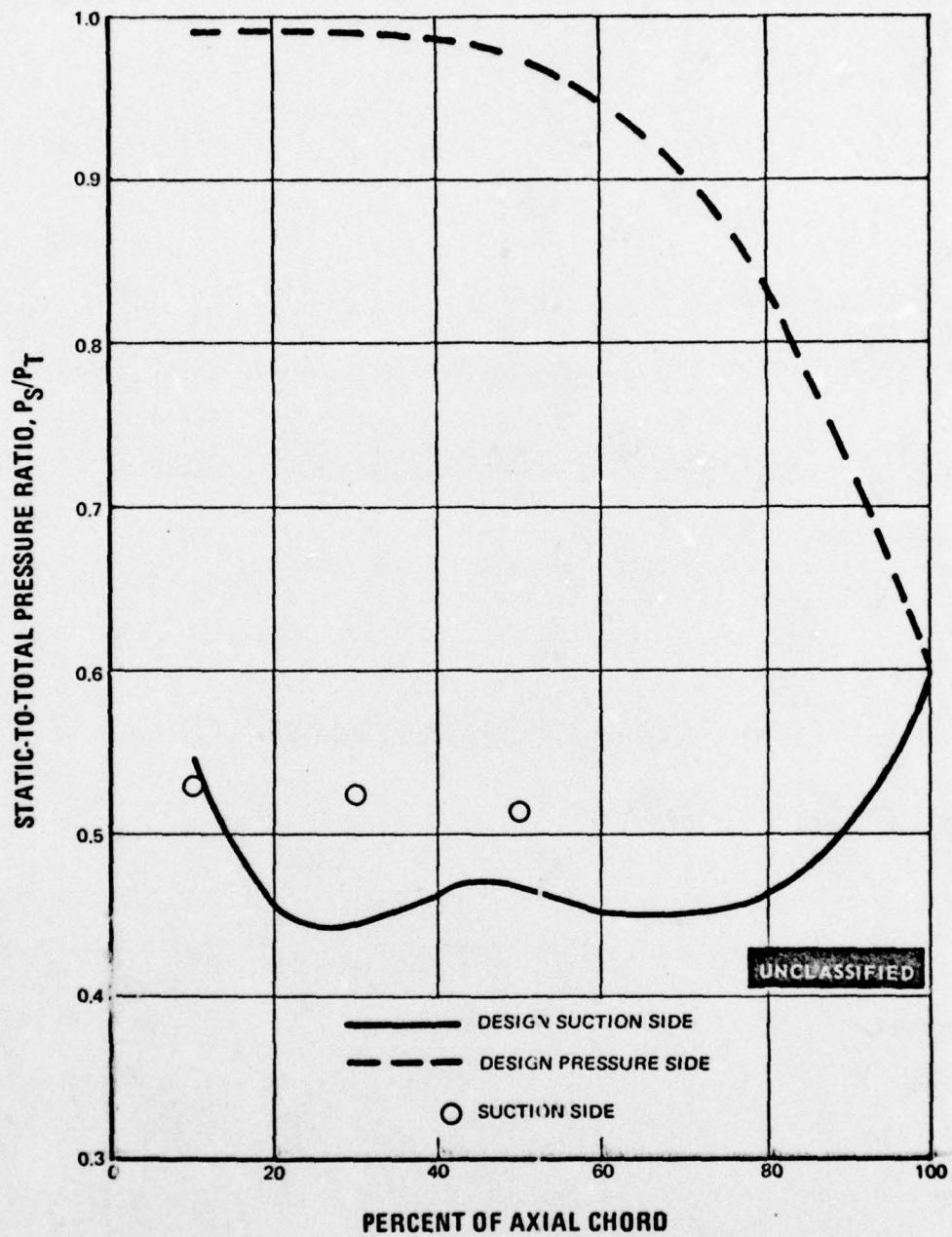


Figure 63 Static-to-Total Pressure Ratio Versus Percent Axial Chord, Second Blade,  
Medium Solidity, Mean Section

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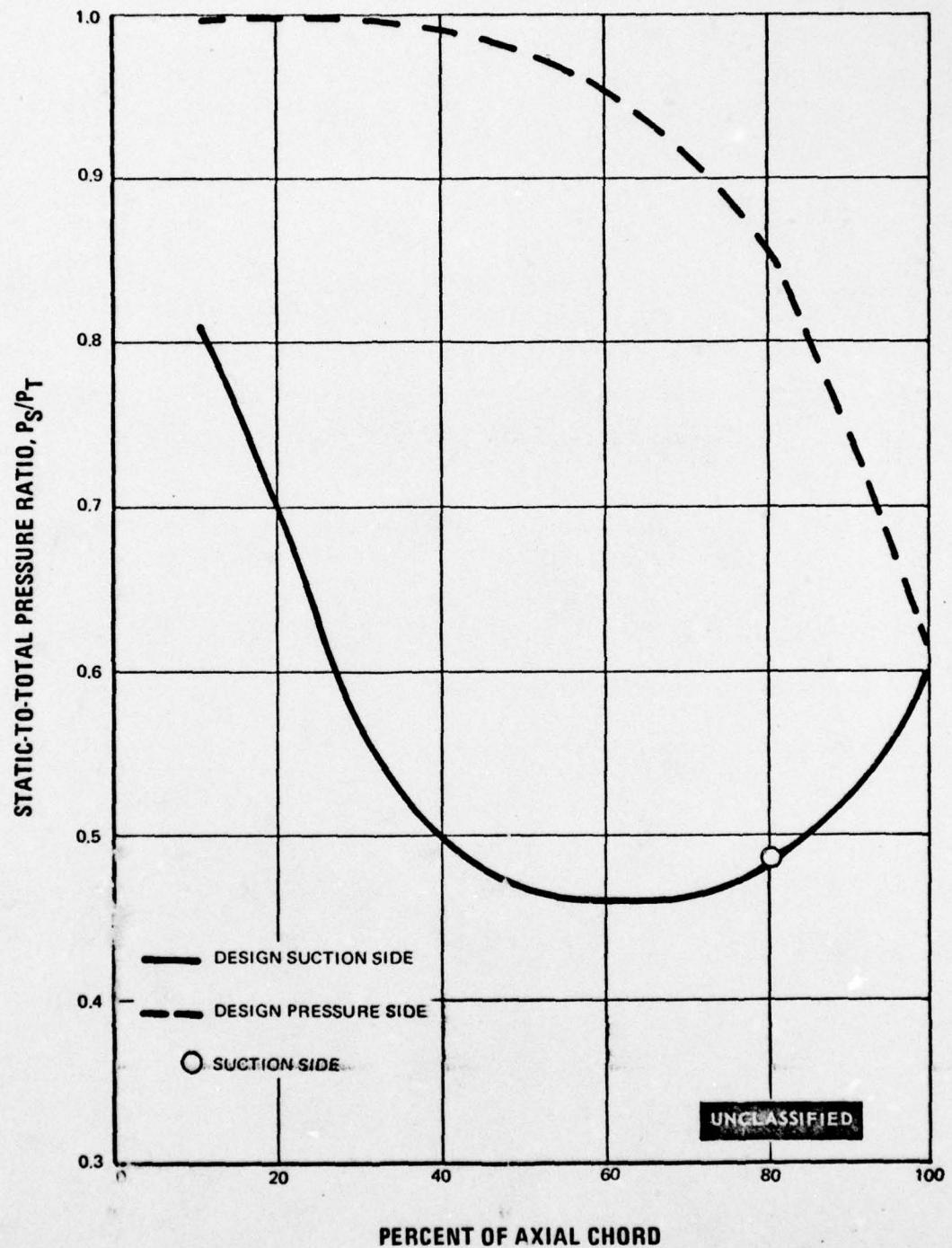


Figure 64      Static-to-Total Pressure Ratio Versus Percent Axial Chord, Second Blade,  
Medium Solidity, Tip Section

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(U) A complete summary of the loss data taken on medium solidity airfoils for the various midspan Mach numbers and Reynolds numbers is shown in Table VI.

TABLE VI  
SUMMARY OF TASK II<sup>d</sup> DATA - MEDIUM SOLIDITY AIRFOILS

Airfoil	Midspan Turbine Design $M_2$	Midspan Test $M_2$	Midspan Test Reynolds No. *Design	Midspan Test $1-\phi^2$	Flow Weighted Averages		
					Root To Mean Test $1-\phi^2$	Mean To Tip Test $1-\phi^2$	Overall Test $1-\phi^2$
First Vane	0.852	0.837	$*4.38 \times 10^5$	0.031	0.051	0.050	0.051
		0.739	$4.38 \times 10^5$	0.035	0.054	0.055	0.054
		0.941	$4.38 \times 10^5$	0.031	0.040	0.044	0.042
		0.827	$6.56 \times 10^5$	0.040	0.072	0.059	0.065
		0.814	$2.19 \times 10^5$	0.047	0.079	0.068	0.073
		0.880	$3.78 \times 10^5$	0.026	—	—	—
First Blade	0.788	0.774	$*2.26 \times 10^5$	0.114	0.200	0.125	0.159
		0.671	$2.26 \times 10^5$	0.120	0.191	0.145	0.167
		0.867	$2.26 \times 10^5$	0.091	0.148	0.114	0.129
Second Vane	0.870	0.818	$*2.95 \times 10^5$	0.123	0.085	0.214	0.157
		0.732	$2.95 \times 10^5$	0.136	0.078	0.210	0.152
		0.915	$2.95 \times 10^5$	0.121	0.86	0.202	0.150
Second Blade	0.904	0.935	$*1.52 \times 10^5$	0.033	0.042	0.092	0.070
		0.839	$1.52 \times 10^5$	0.033	0.037	0.103	0.075
		1.000	$1.52 \times 10^5$	0.046	0.058	0.093	0.078

## 5. SUMMARY

(U) A comparison of the performance of the normal and medium solidity airfoils is included in this Section.

(U) The basic objective of Task II<sup>d</sup> was to evaluate the performance of cascades of lower solidity and higher load coefficient designed for similar aerodynamic inlet and exit conditions as the baseline normal solidity airfoils. With the completion of the medium solidity performance evaluation, a comparison of the performance of these two designs can be made. A summary of the measured and predicted loss coefficients for the normal and medium solidity designs at the turbine design point Mach number and Reynolds number is shown in Table VII. Based on this loss data and on the exit gas angle distributions reported previously, it can be concluded that the normal solidity designs are sufficiently superior to the medium solidity designs when compared under annular segment cascade conditions to justify their choice in the final turbine design.

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TABLE VII

## SUMMARY OF DESIGN POINT 1- $\phi^2$ LOSS COEFFICIENTS FOR NORMAL AND MEDIUM SOLIDITY AIRFOILS

	Turbine Design Midspan Exit Mach No.	Midspan		Root To Mean Average Test	Mean To Tip Average Test	Overall Average Test
		Test	Predicted For Fully Turbulent Boundary Layer			
Normal Solidity First Vane	0.854	0.017 0.023*	0.031	0.029	0.035	0.034 0.032*
Medium Solidity First Vane	0.854	0.031	0.035	0.051	0.050	0.051
Normal Solidity First Blade	0.780	0.0266	0.049	0.0368	0.0433	0.040
Medium Solidity First Blade	0.780	0.114	0.043	0.200	0.125	0.159
Normal Solidity Second Vane	0.869	0.021 0.028*	0.034	0.0275	0.0325	0.030 0.034*
Medium Solidity Second Vane	0.869	0.123	0.034	0.085	0.214	0.157
Normal Solidity Second Blade	0.904	0.028	0.040	0.0322	0.0438	0.038
Medium Solidity Second Blade	0.904	0.033	0.046	0.042	0.092	0.070

\* Inlet-Screen Installed

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(U) However, due to the transitional nature of the airfoil, evidenced by the very low profile losses of the normal solidity airfoils and the apparent importance of the effective free stream turbulence level shown by the inlet screen tests, it is not possible to make a definite conclusion concerning the relative performance of these designs in a rotating turbine. If the effective free stream turbulence level in the rotating turbine is sufficiently high to move the boundary layer transition further toward the airfoil leading edge on the suction side, it is expected that the profile losses on the normal solidity airfoils will approach the higher fully turbulent boundary layer predictions. For the medium solidity airfoils, this change in boundary layer transition could actually improve the performance, since a more turbulent boundary layer would have greater resistance to separation.

(U) It is recommended that the tests of the medium solidity airfoils are repeated under conditions for which the boundary layer is certain to be turbulent. Experience shows that the surest way to achieve this goal is by use of a suction surface trip wire. Such tests would not only be especially informative for this specific program, but would be of interest to all aerodynamic programs.

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## SECTION V BOUNDARY LAYER CONTROL OPTIMIZATION (TASK III a)

### 1. OBJECTIVE

(U) Determine the optimum geometry for the selected boundary layer control method.

### 2. TASK OBJECTIVE

(U) Three geometrical variations of the selected boundary layer control scheme were evaluated in an annular segment cascade on the second vane baseline airfoil. The purpose of these tests was to further investigate techniques for reducing end-wall losses. The selected scheme was local airfoil recambering, since after an experimental evaluation of (1) local recontouring (2) local recambering (3) end-wall contouring and (4) use of flow fences and application of increased surface roughness (Reference 4, Section V), this method proved to be most promising to diminish the end-wall losses.

(U) Each airfoil design was separately tested at the design incidence angle and Reynolds number and three Mach numbers near the design point. Exit surveys were made over the entire central passage containing the test airfoil in order to obtain loss coefficients and exit angle deviations. During these tests, oil and graphite flow visualization techniques were used to observe flow patterns on the airfoils.

(U) The task objectives were met and the geometry for the locally recambered airfoil which had the lowest end-wall losses was determined by:

- Reconstruction of the entire exit plane loss distribution for each scheme
- Reconstruction of the entire exit plane flow pattern for each scheme

(U) Analysis of this exit plane data revealed the high and low loss regions in the passages and the merits of each scheme were determined.

### 3. CASCADE AIRFOIL DESIGN

(U) Three geometrical variations of local recambering were applied to the second vane baseline airfoil. The design exit angle distributions are shown in Figure 65 for the three variations referred to as A, B, and C. The total integrated spanwise work was maintained constant at the baseline value for each recambering design. Included for comparison (Figure 65) are the exit gas angle distributions for the original baseline airfoil and the first local recambering (Task II c). The airfoil elevations of recambering designs A, B and C are presented in Figures 66, 67 and 68. The airfoil sections are presented in Figures 69 through 91, and the corresponding predicted airfoil surface pressure distributions, convergence and radius of curvature are presented in Figures 92 through 106. The fabrication coordinates for each of the sections for these three airfoils are

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tabulated in the Appendix. For recambering design A they appear in Tables VIII through XV, for recambering design B in Tables XVI through XXIII and for recambering design C in Tables XXIV through XXXI. Such additional airfoil information as airfoil angles, gaging distance, airfoil area, axial chord, and uncovered turning is also tabulated therein.

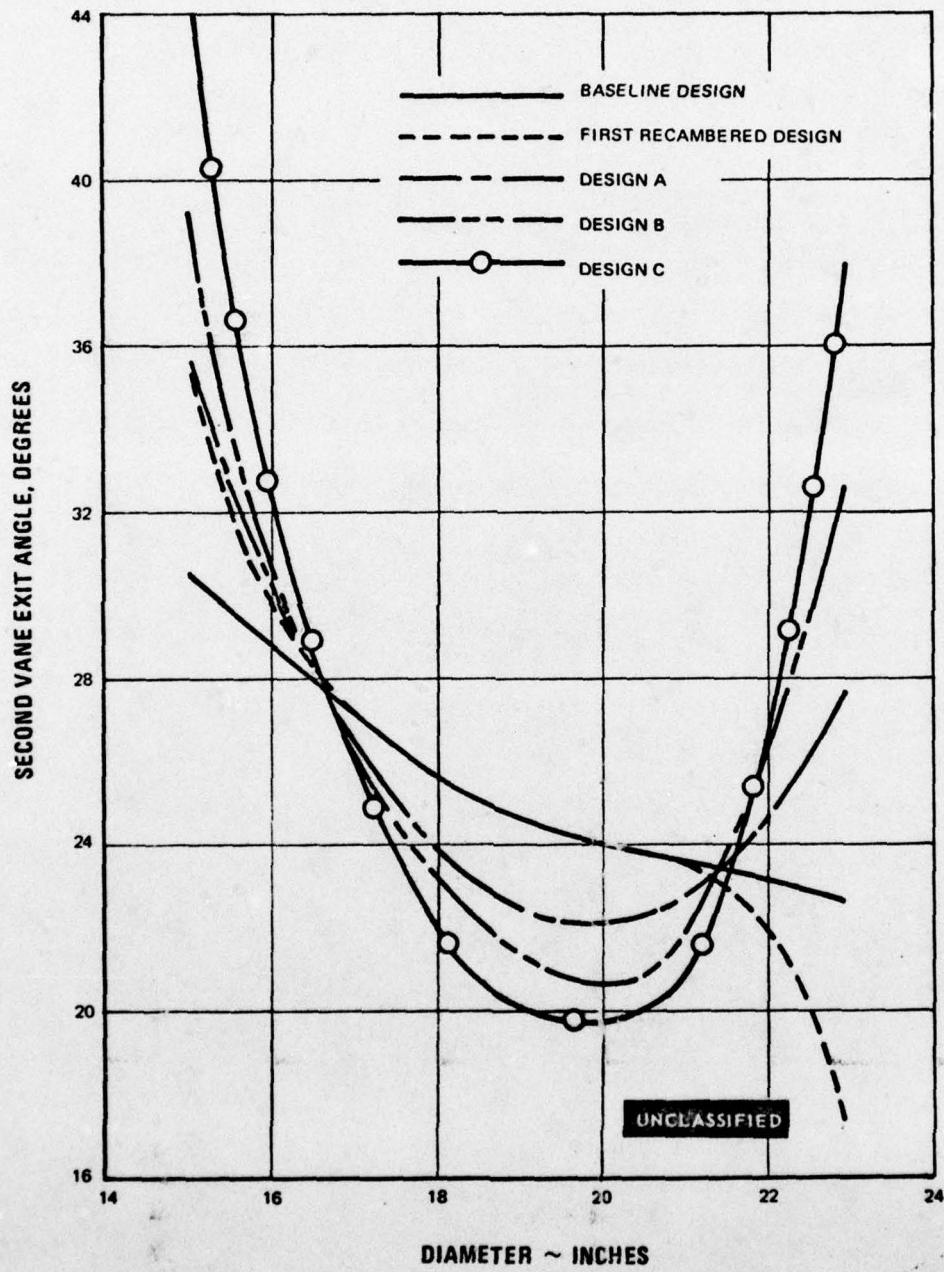


Figure 65 Second Vane Recambering Design Comparison

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NOTE: ACTUAL AIRFOIL SHOULD BE EXTENDED TO R=11.945" (TIP EXTENDED SECTION, K-K) AND R=7.0" (ROOT EXTENDED SECTION, J-J). THE X AND Y DEFINING COORDINATES FOR THE TIP AND ROOT EXTENDED SECTIONS ARE THE SAME AS SECTIONS GG AND FF RESPECTIVELY.

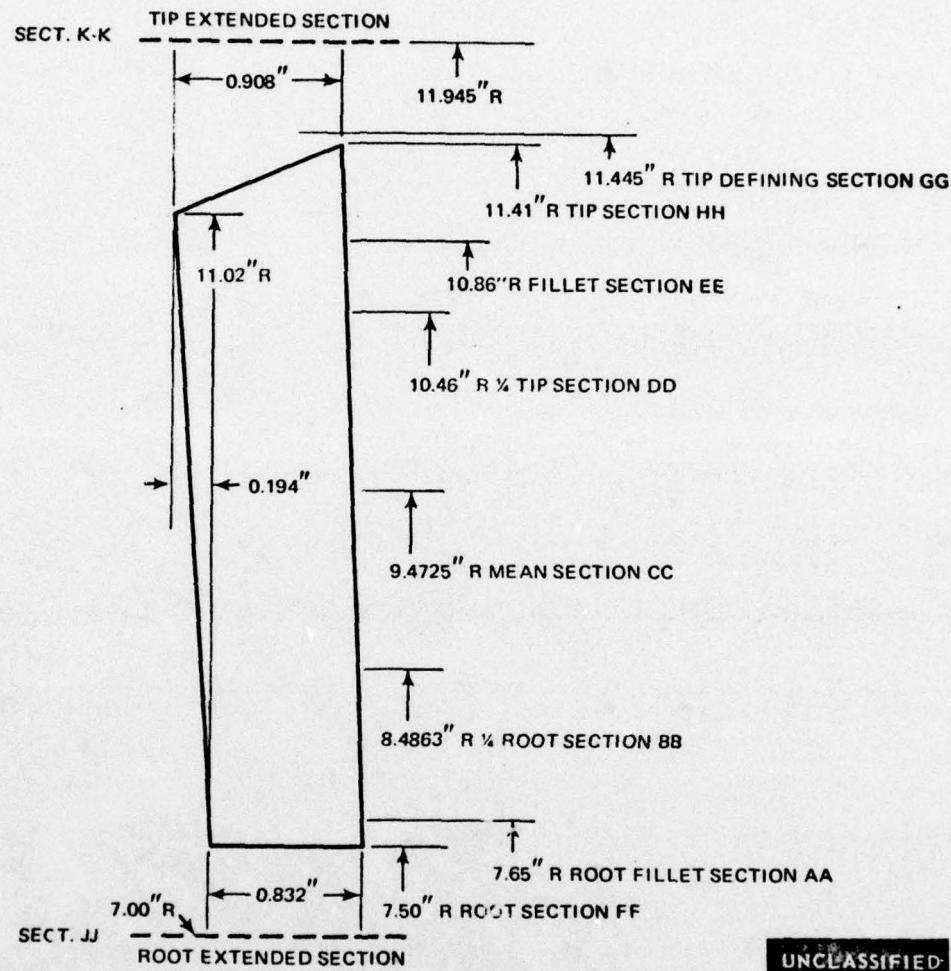


Figure 66 Elevation and Section Location of Second Vane, Recambering Design A

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NOTE: CONSTANT SECTION EXISTS BETWEEN R=11.445 (TIP DEFINING SECTION) AND R=11.945 (TIP EXTENDED SECTION). CONSTANT SECTION EXISTS BETWEEN R=7.5 (ROOT SECTION) AND R=7.0 (ROOT EXTENDED SECTION).

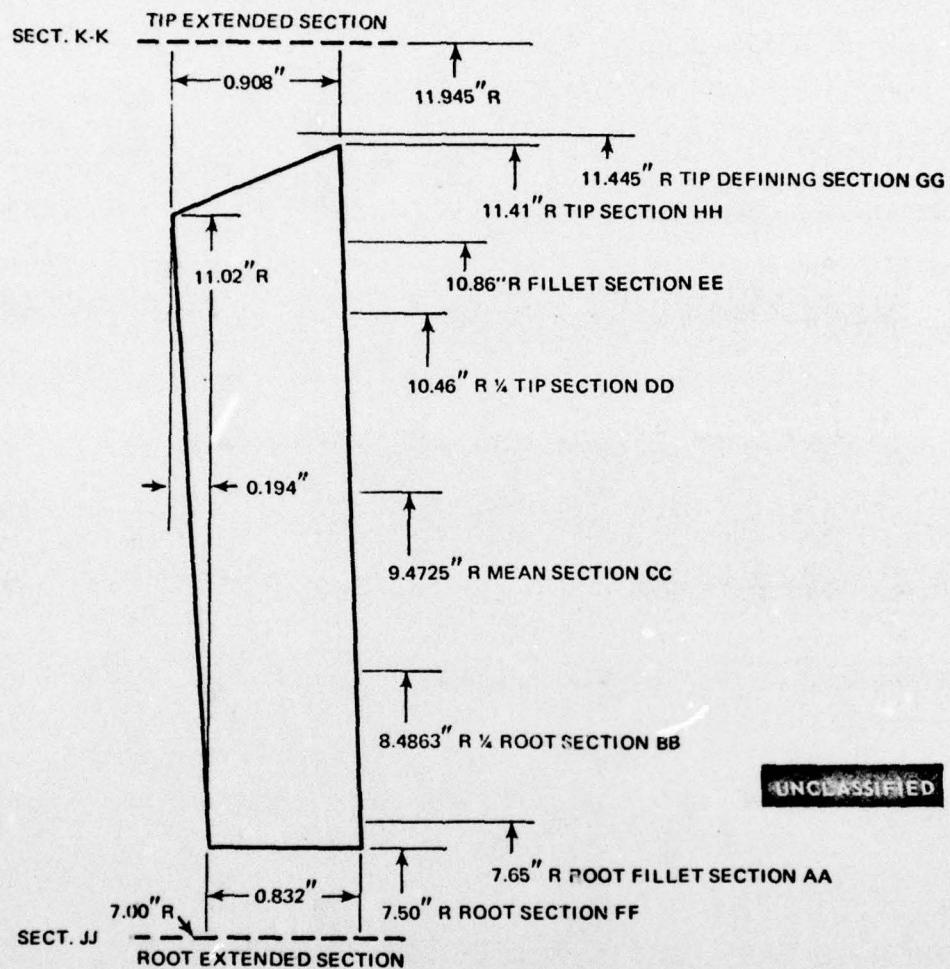


Figure 67 Elevation and Section Location of Second Vane, Recambering Design B

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NOTE: CONSTANT SECTION EXISTS BETWEEN R=11.445 (TIP DEFINING SECTION) AND R=11.945 (TIP EXTENDED SECTION). CONSTANT SECTION EXISTS BETWEEN R=7.5 (ROOT SECTION) AND R=7.0 (ROOT EXTENDED SECTION).

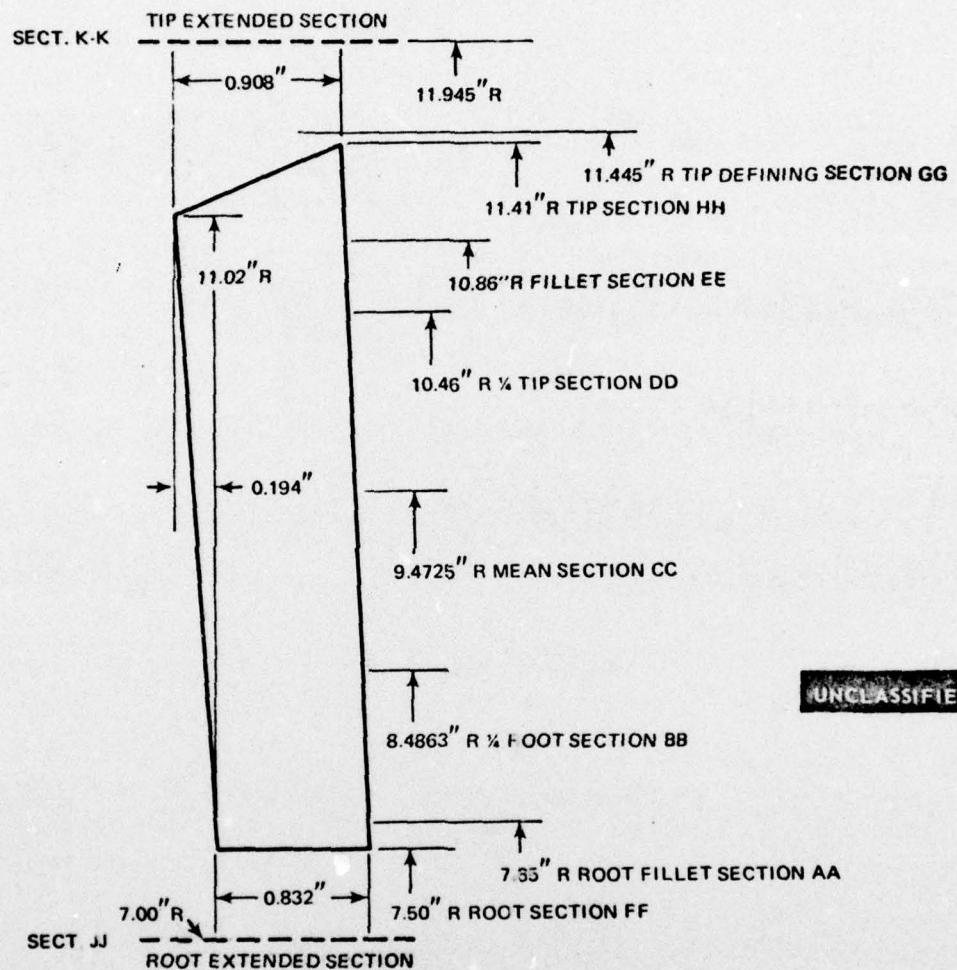
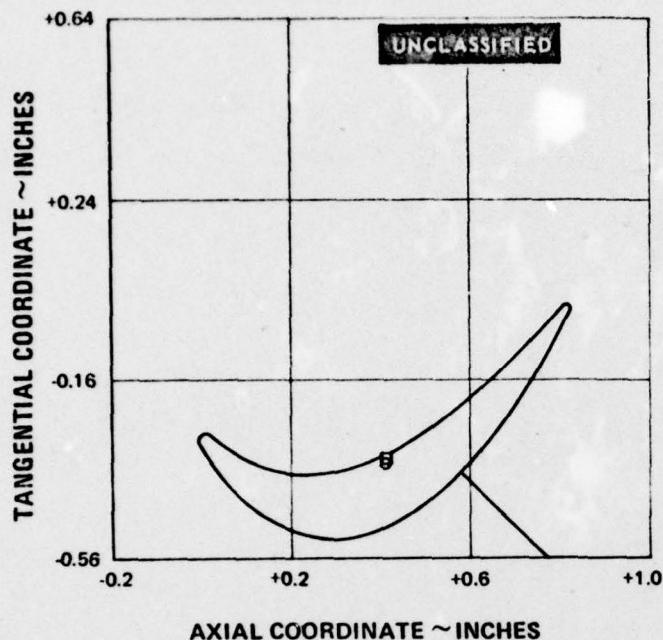


Figure 68 Elevation and Section Location of Second Vane, Recamering Design C

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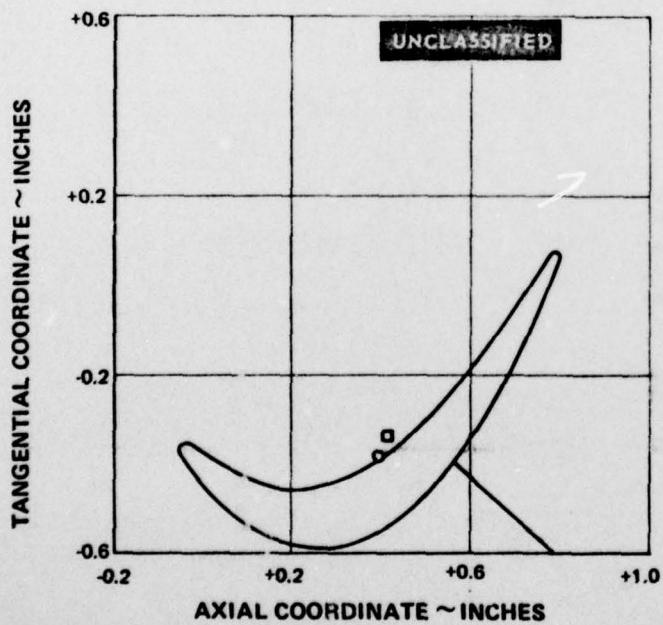


**HOT DIMENSIONS**

NUMBER OF FOILS	80.00
DIAMETER	15.3000 IN.
GAGING	0.3240 IN.
PITCH	0.6008 IN.
AXIAL WIDTH	0.8357 IN.
BLADE INLET ANGLE	37.73 DEG.
GAS INLET ANGLE	38.00 DEG.
BLADE EXIT ANGLE	33.83 DEG.
GAS EXIT ANGLE	33.70 DEG.
GAGING ANGLE	32.63 DEG.
UNCOVERED TURNING	17.54 DEG.
PRINCIPAL AXIS	-24.29 DEG.
L. E. RADIUS	0.0179 IN.
T. E. RADIUS	0.0100 IN.
METAL AREA	0.1008 IN. <sup>2</sup>

	X	Y
O C.G.	0.4107	-0.3436
□ RADIAL REF.	0.4132	-0.3347
GAGE	0.5788	-0.3650

Figure 69 Recambering Design A, Second Vane, Root Fillet Section (AA)



**HOT DIMENSIONS**

NUMBER OF FOILS	80.00
DIAMETER	16.9726 IN.
GAGING	0.2969 IN.
PITCH	0.6665 IN.
AXIAL WIDTH	0.8565 IN.
BLADE INLET ANGLE	42.10 DEG
GAS INLET ANGLE	42.10 DEG
BLADE EXIT ANGLE	26.45 DEG
GAS EXIT ANGLE	26.45 DEG
GAGING ANGLE	26.45 DEG
UNCOVERED TURNING	18.49 DEG
PRINCIPAL AXIS	-32.60 DEG
L. E. RADIUS	0.0200 IN.
T. E. RADIUS	0.0100 IN.
METAL AREA	0.1073 IN. <sup>2</sup>

	X	Y
O C. G.	0.3939	-0.3812
□ RADIAL REF.	0.4132	-0.3347
GAGE	0.5531	-0.4044

Figure 70 Recambering Design A, Second Vane, 1/4 Root Section (BB)

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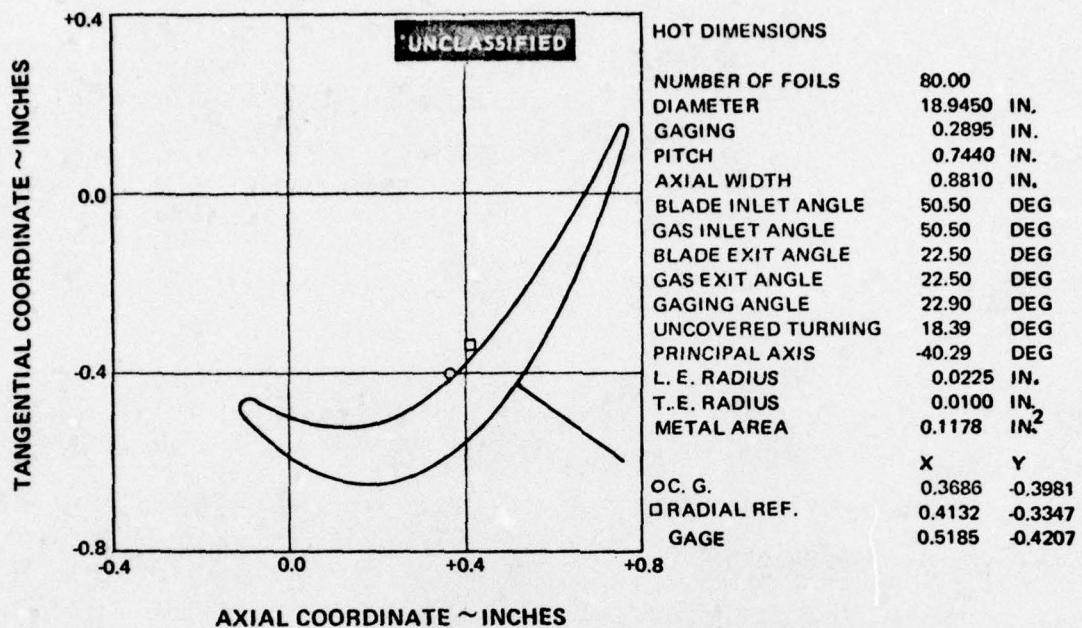


Figure 71 Recambering Design A, Second Vane, Mean Section (CC)

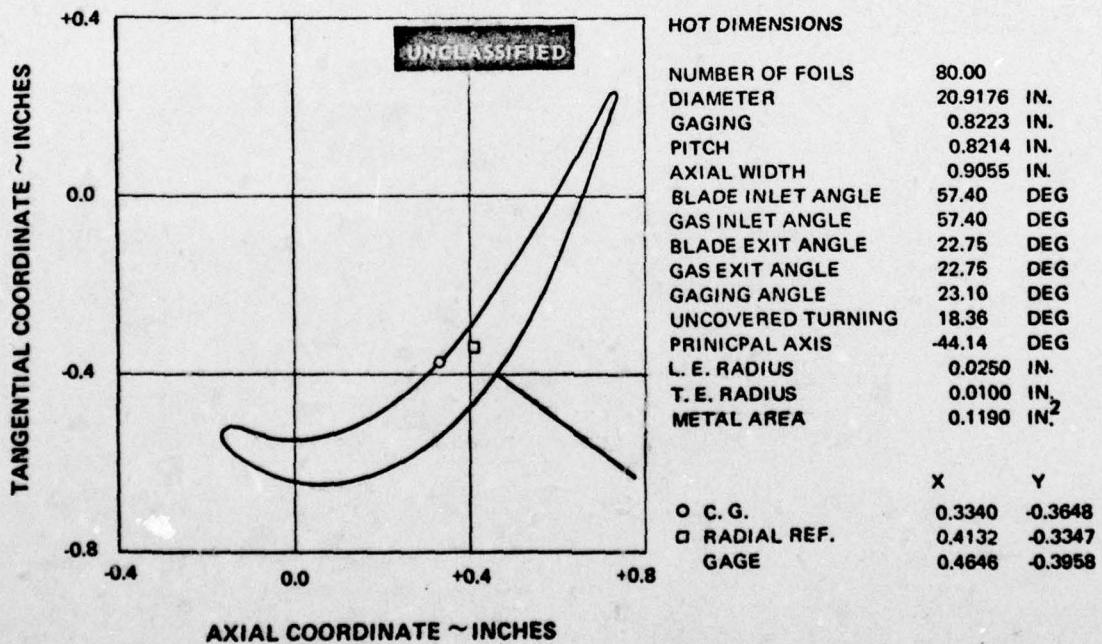


Figure 72 Recambering Design A, Second Vane, 1/4 Tip Section (DD)

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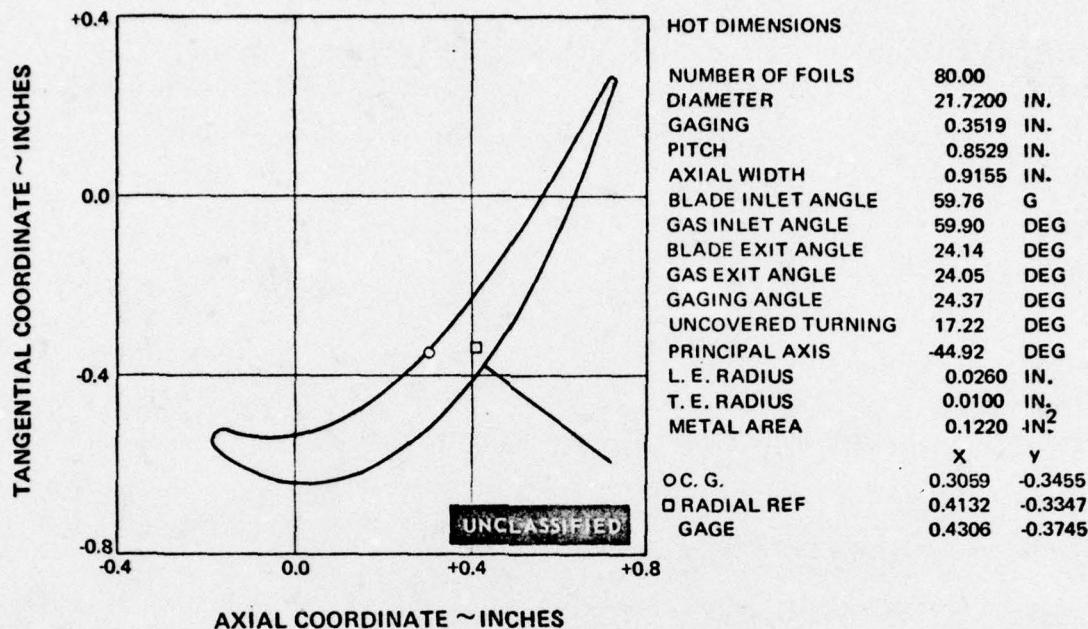


Figure 73 Recambering Design A, Second Vane, Fillet Section (EE)

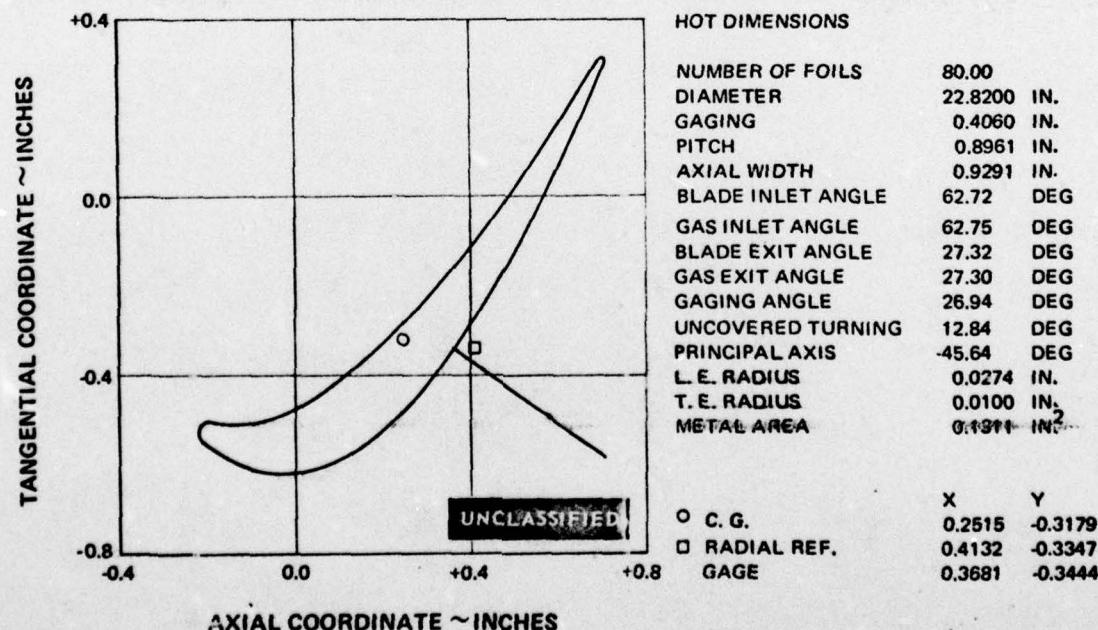


Figure 74 Recambering Design A, Second Vane, Tip Section (HH)

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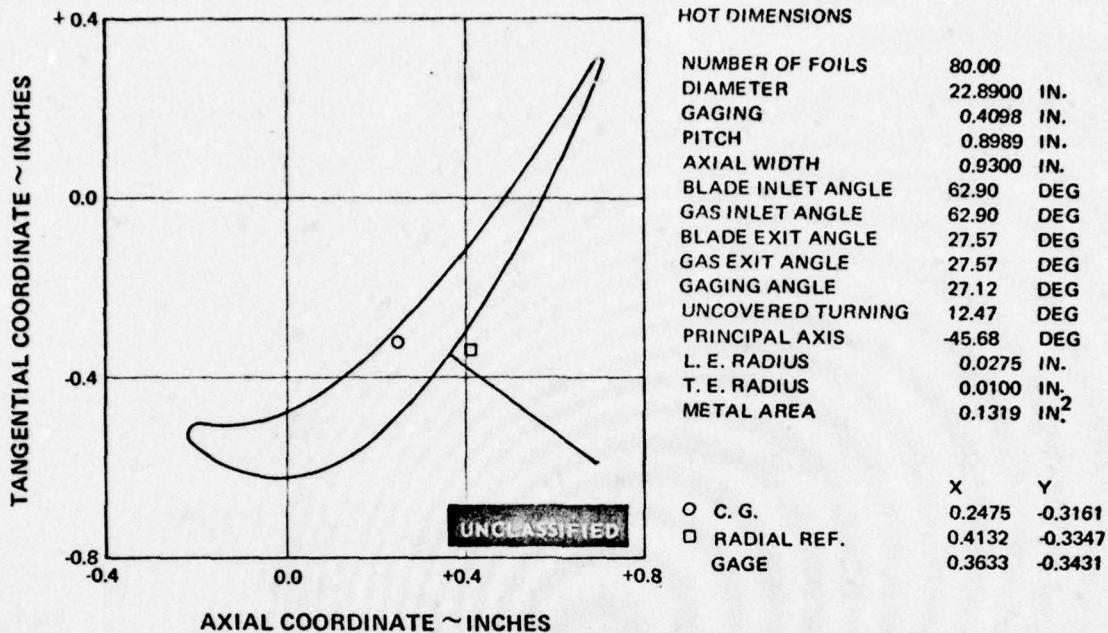


Figure 75 Recambering Design A, Second Vane, Tip Defining Section (GG)

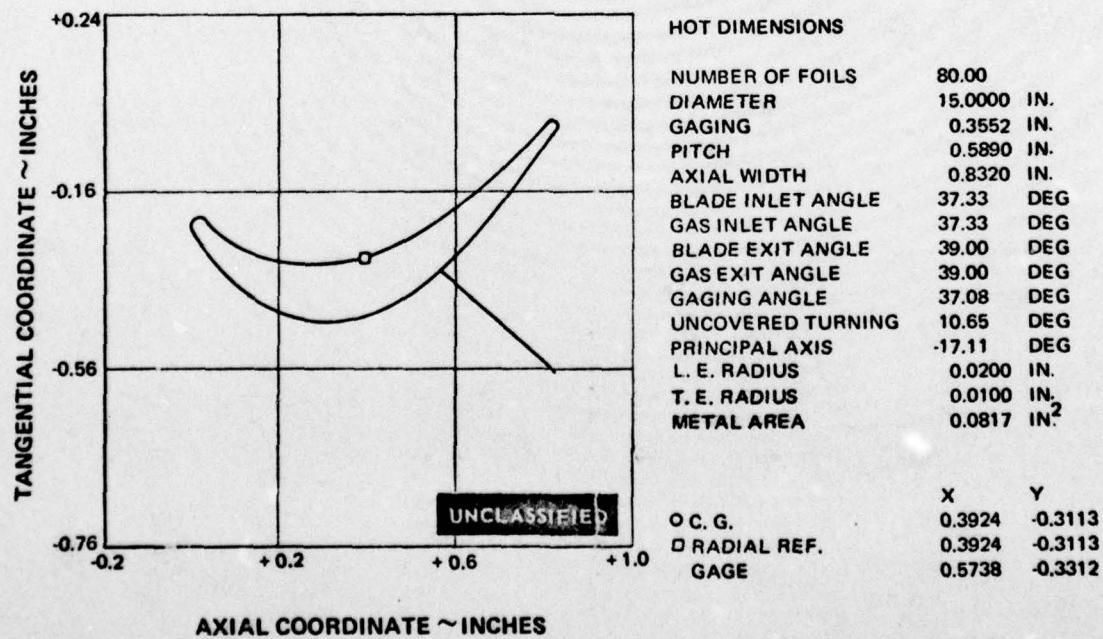


Figure 76 Recambering Design B, Second Vane, Root Section (FF)

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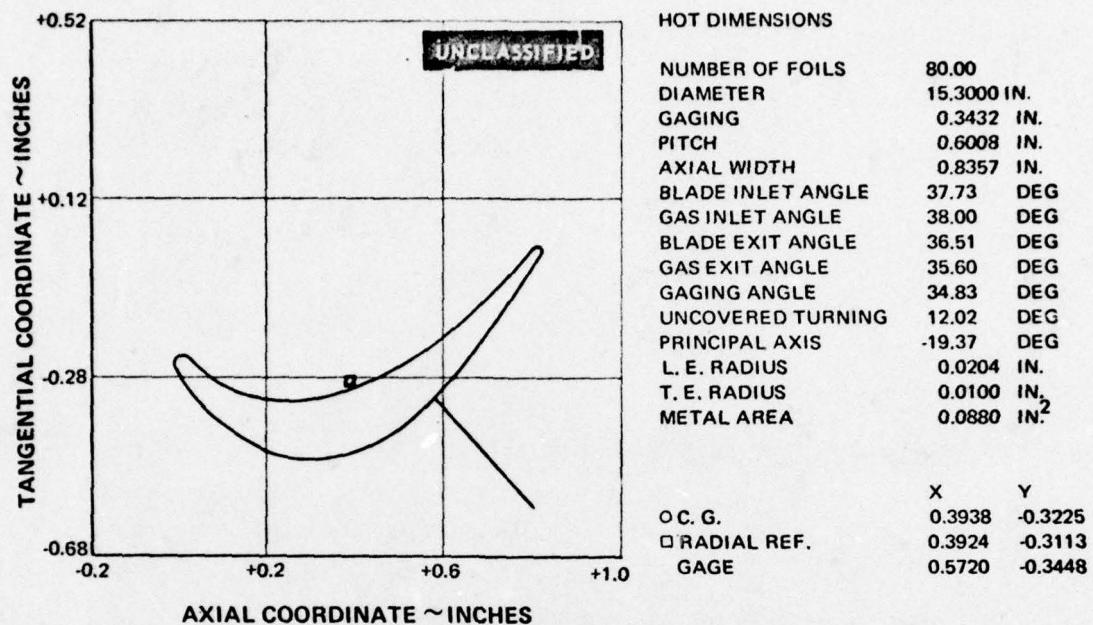


Figure 77 Recambering Design B, Second Vane, Root Fillet Section (AA)

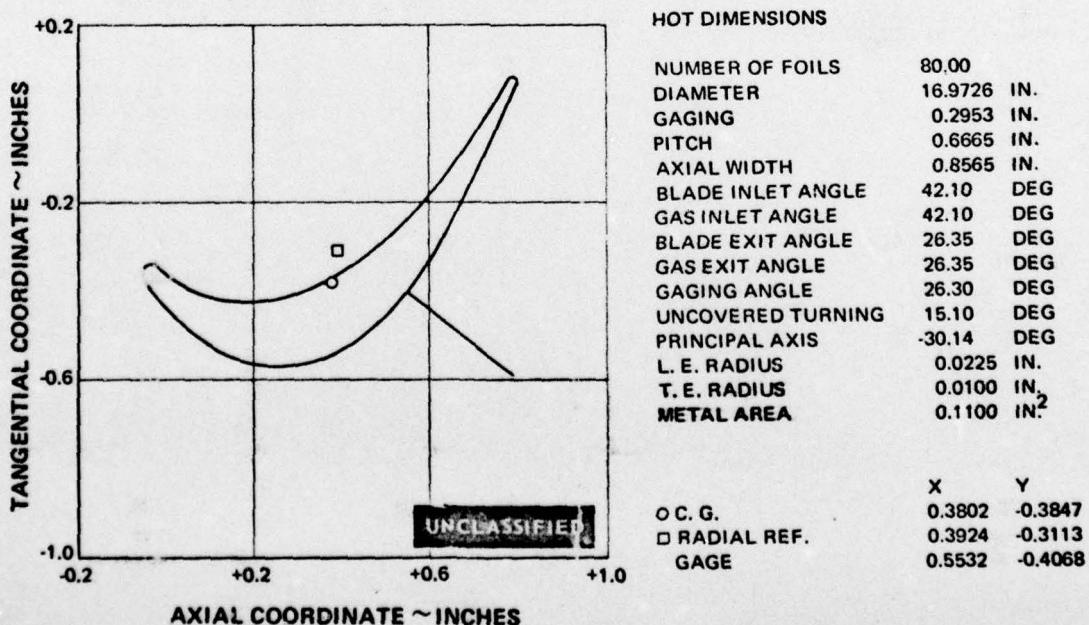


Figure 78 Recambering Design B, Second Vane, 1/4 Root Section (BB)

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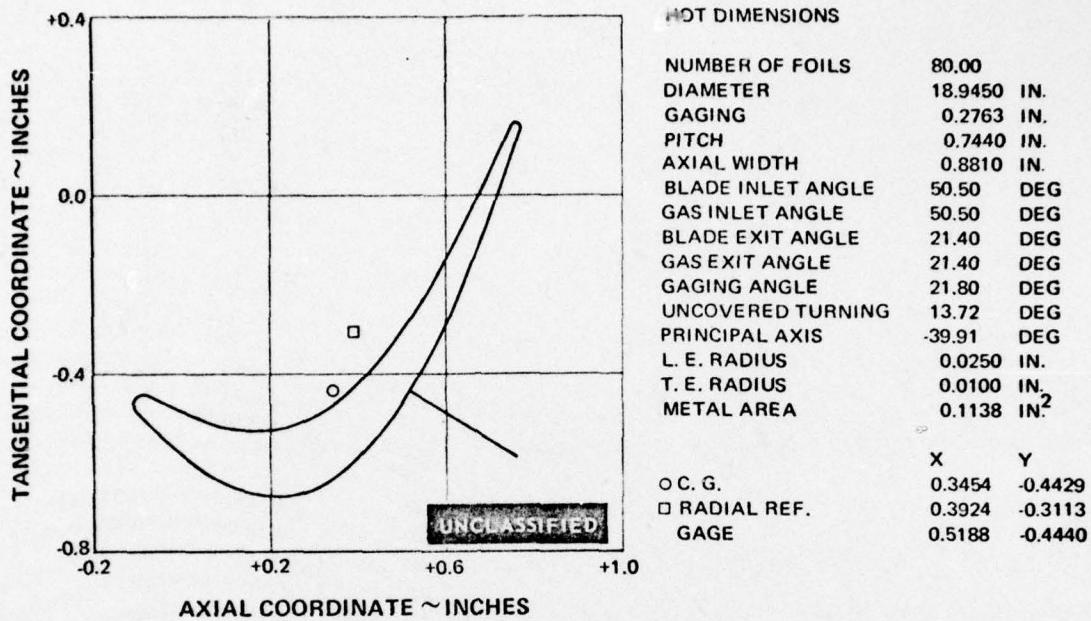


Figure 79 Recambering Design B, Second Vane, Mean Section (CC)

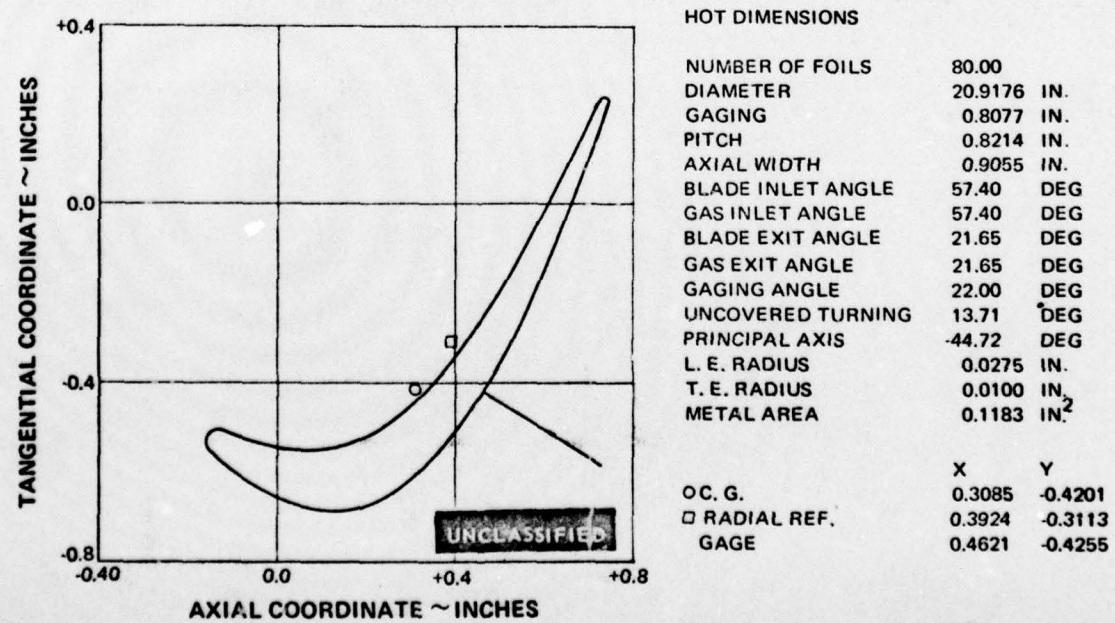
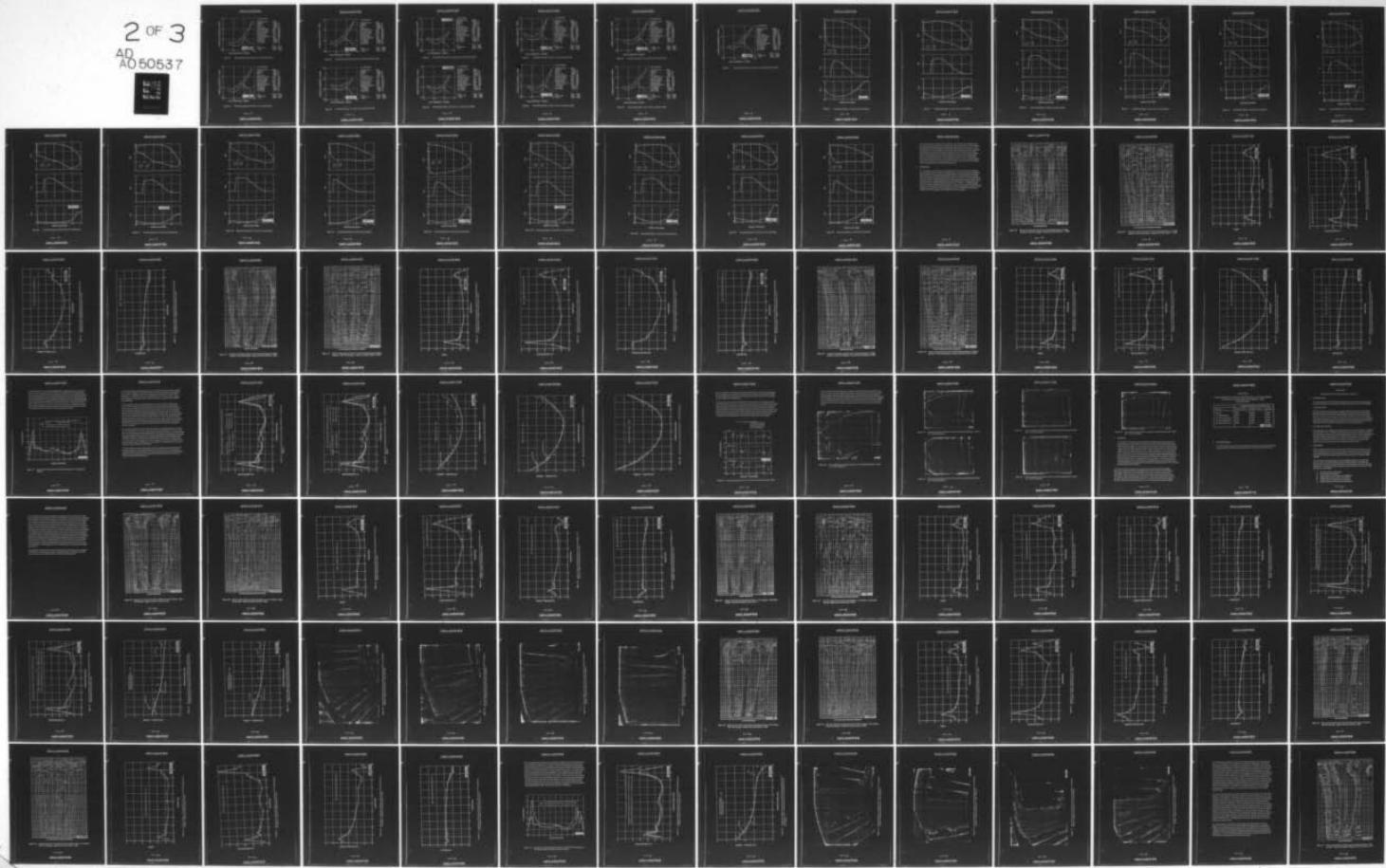


Figure 80 Recambering Design B, Second Vane, 1/4 Tip Section (DD)

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INVESTIGATION OF A HIGHLY LOADED TWO-STAGE FAN-DRIVE TURBINE. V--ETC(U)  
JUN 70 M WELNA, D E DAHLBERG, W H HEISER F33615-68-C-1208  
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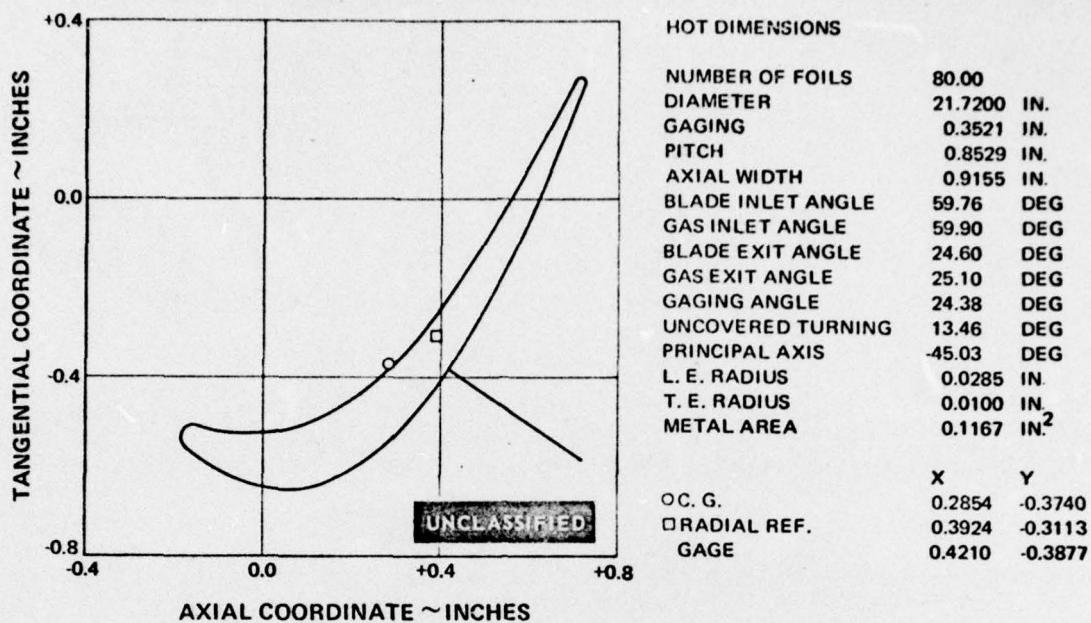


Figure 81 Recambering Design B, Second Vane, Fillet Section (EE)

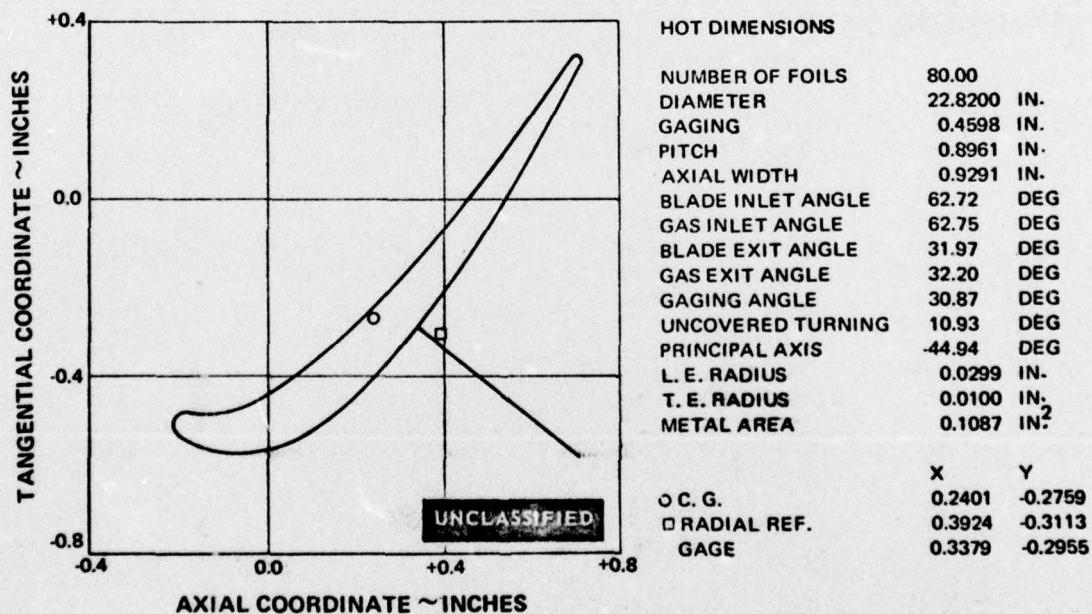
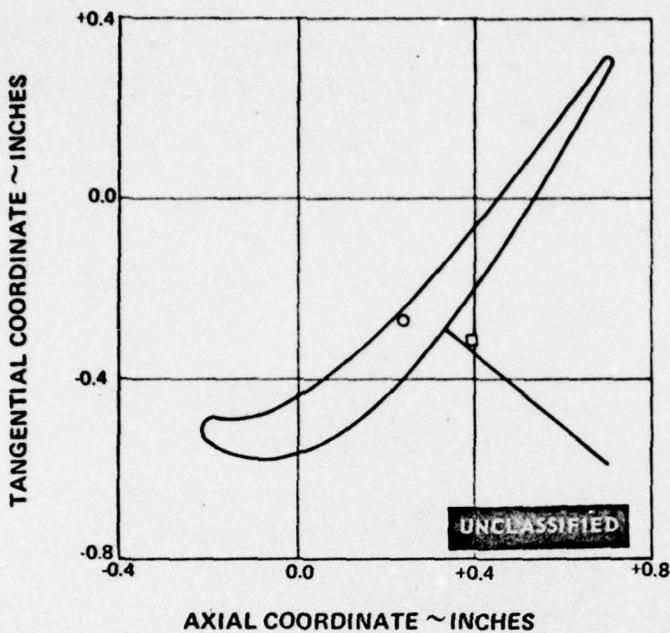


Figure 82 Recambering Design B, Second Vane, Tip Section (HH)

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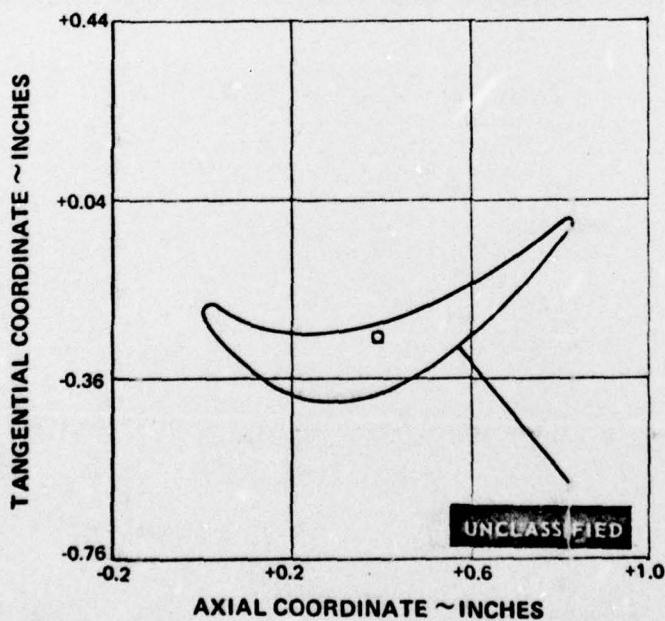
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## HOT DIMENSIONS

NUMBER OF FOILS	80.00
DIAMETER	22.8900 IN.
GAGING	0.4687 IN.
PITCH	0.8989 IN.
AXIAL WIDTH	0.9300 IN.
BLADE INLET ANGLE	62.90 DEG
GAS INLET ANGLE	62.90 DEG
BLADE EXIT ANGLE	32.57 DEG
GAS EXIT ANGLE	32.57 DEG
GAGING ANGLE	31.43 DEG
UNCOVERED TURNING	10.58 DEG
PRINCIPAL AXIS	-44.96 DEG
L. E. RADIUS	0.0300 IN.
T. E. RADIUS	0.0100 IN.
METAL AREA	0.1079 IN. <sup>2</sup>

	X	Y
O.C. G.	0.2365	-0.2683
□ RADIAL REF.	0.3924	-0.3113
GAGE	0.3312	-0.2881



## HOT DIMENSIONS

NUMBER OF FOILS	80.00
DIAMETER	15.0000 IN.
GAGING	0.3860 IN.
PITCH	0.5890 IN.
AXIAL WIDTH	0.8320 IN.
BLADE INLET ANGLE	37.33 DEG
GAS INLET ANGLE	37.33 DEG
BLADE EXIT ANGLE	44.00 DEG
GAS EXIT ANGLE	44.00 DEG
GAGING ANGLE	40.94 DEG
UNCOVERED TURNING	12.06 DEG
PRINCIPAL AXIS	-16.36 DEG
L. E. RADIUS	0.0200 IN.
T. E. RADIUS	0.0100 IN.
METAL AREA	0.0931 IN. <sup>2</sup>

	X	Y
O.C. G.	0.3919	-0.2691
□ RADIAL REF.	0.3919	-0.2691
GAGE	0.5732	-0.2910

Figure 84 Recambering Design C, Second Vane, Root Section (FF)

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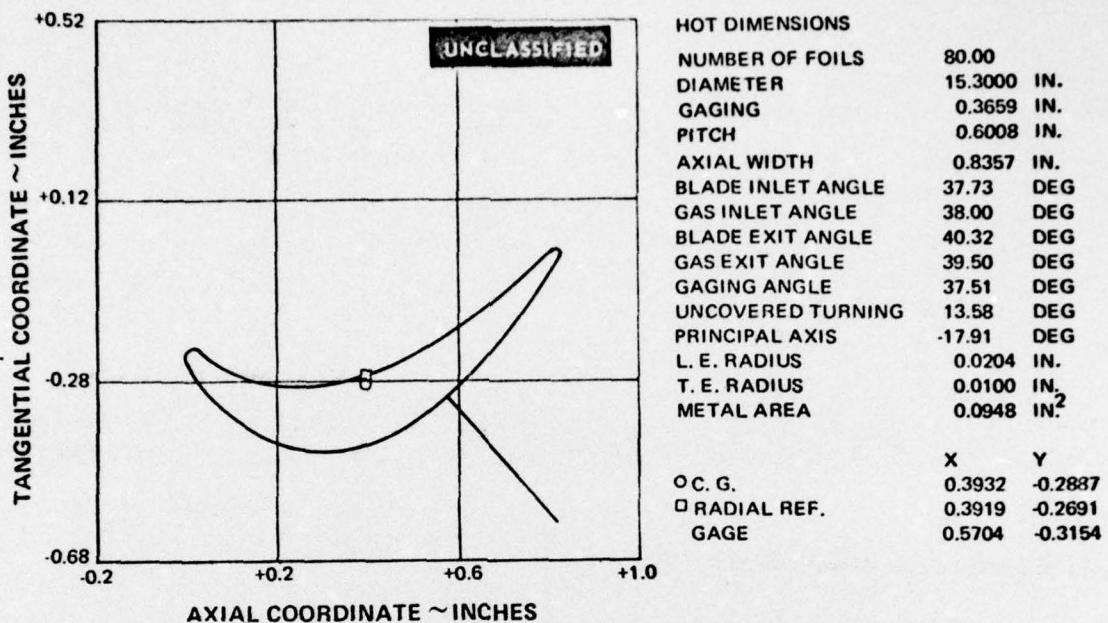


Figure 85 Recambering Design C, Second Vane, Root Fillet Section (AA)

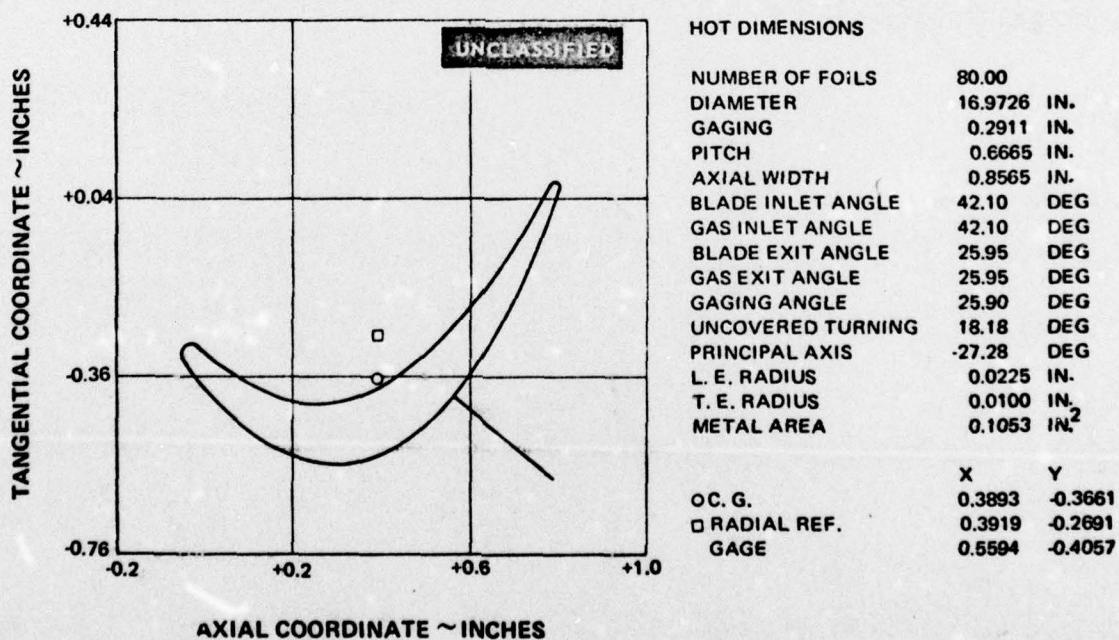


Figure 86 Recambering Design C, Second Vane, 1/4 Root Section (BB)

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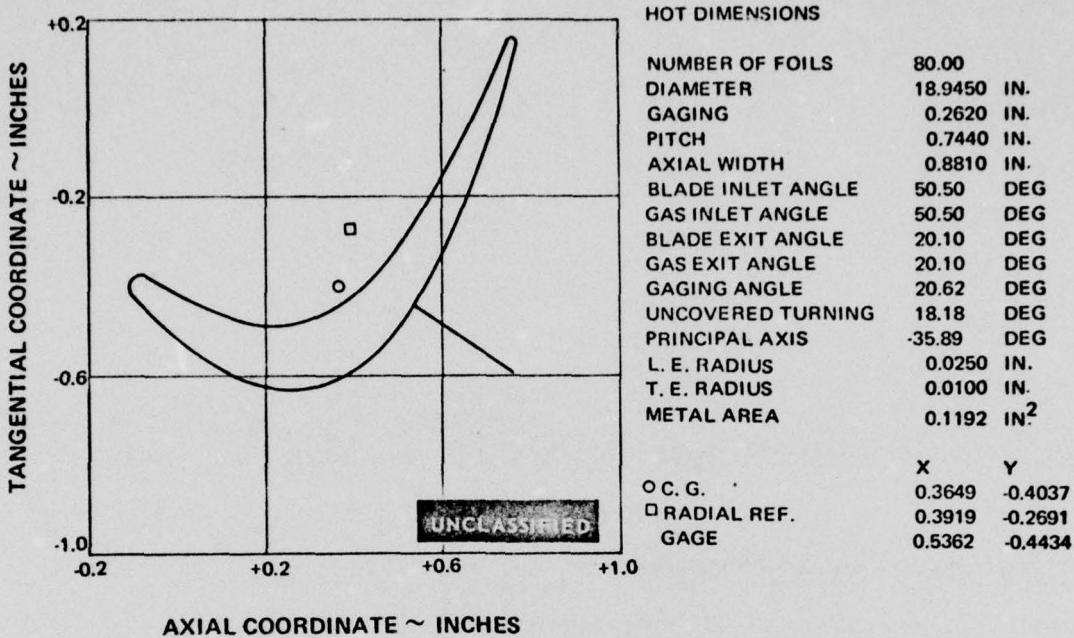


Figure 87 Recambering Design C, Second Vane, Mean Section (CC)

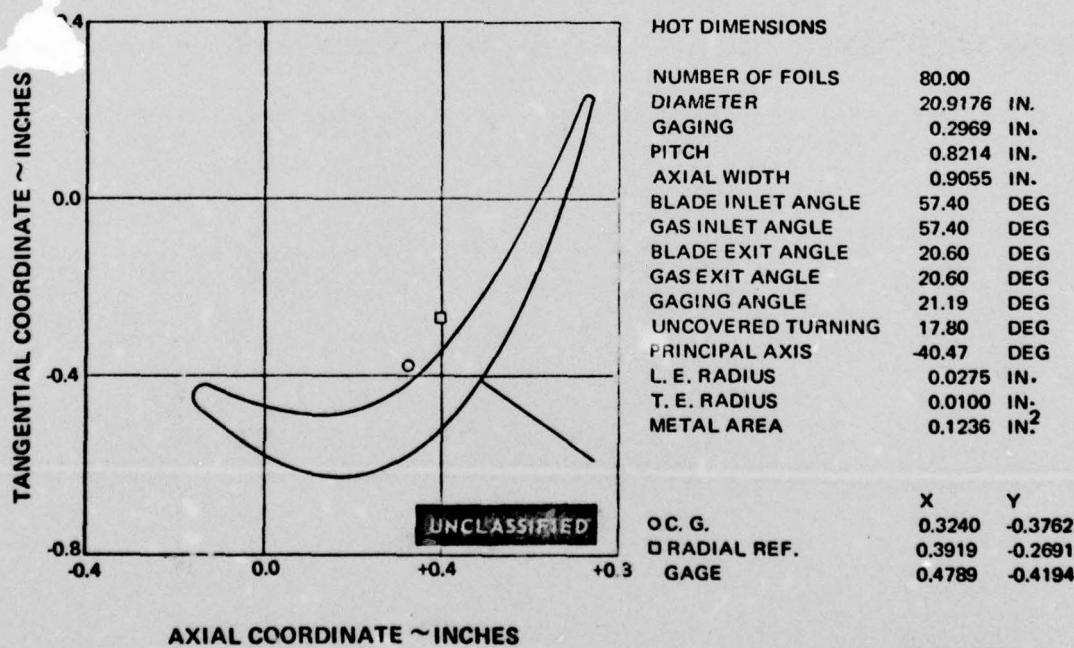


Figure 88 Recambering Design C, Second Vane, 1/4 Tip Section (DD)

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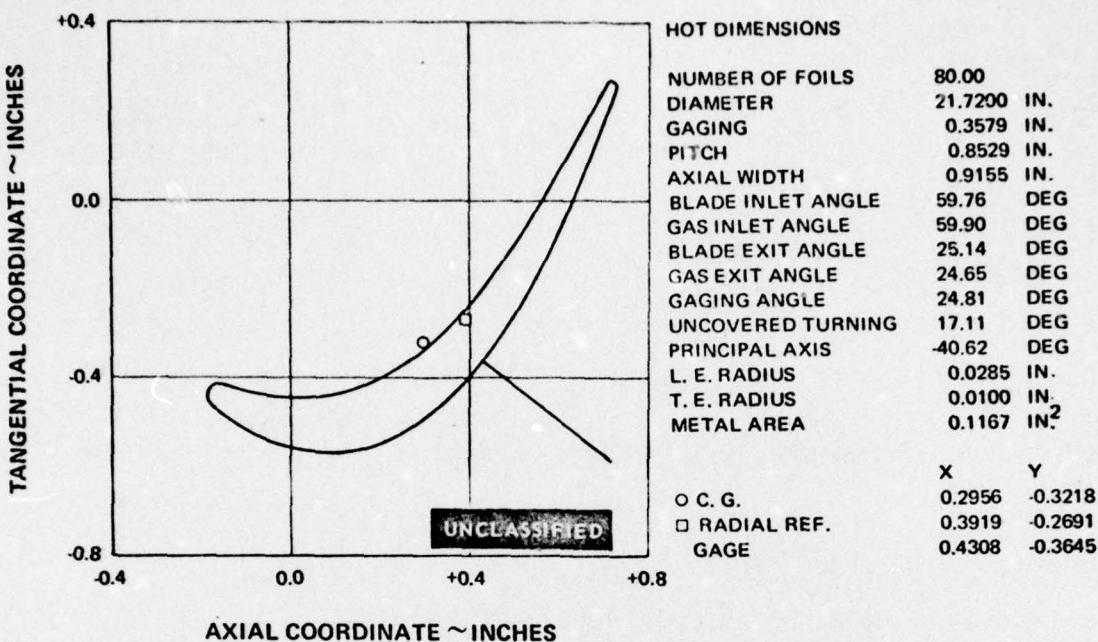


Figure 89 Recambering Design C, Second Vane, Fillet Section (EE)

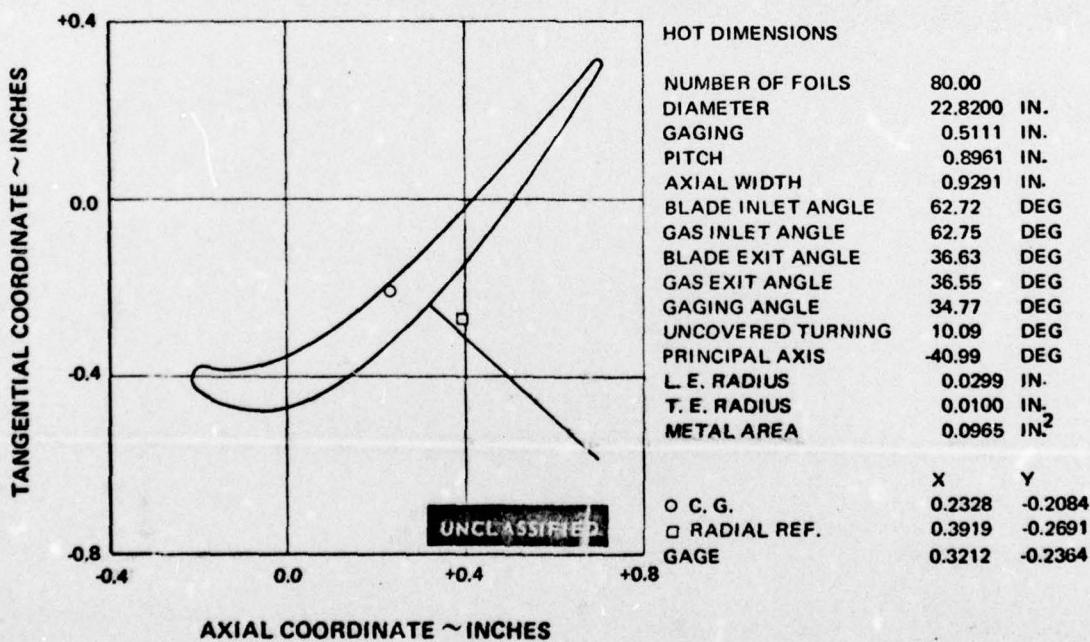


Figure 90 Recambering Design C, Second Vane, Tip Section (HH)

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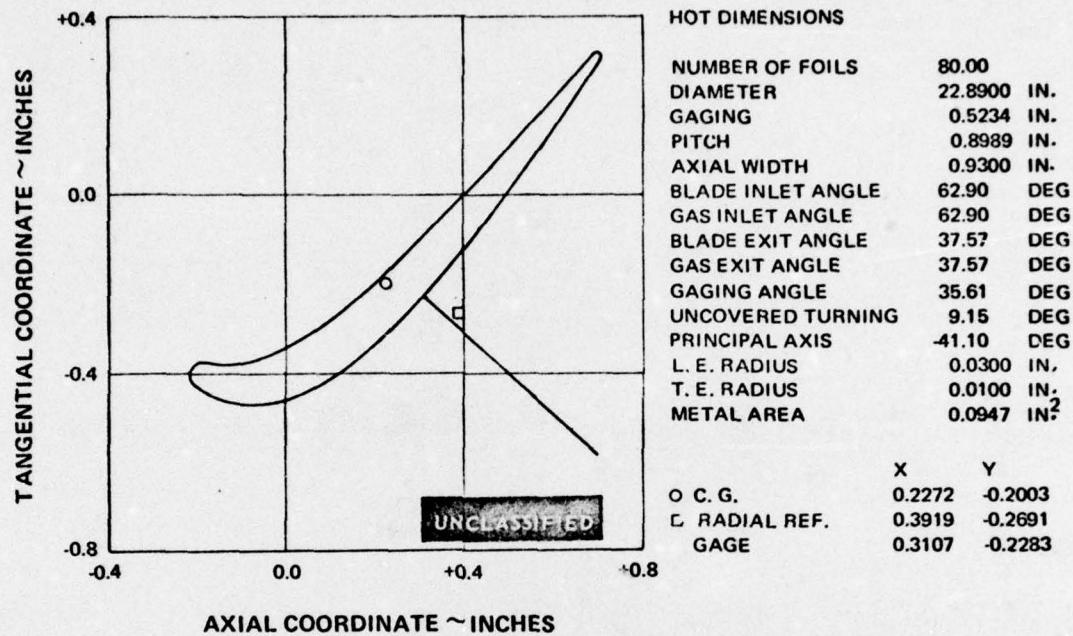


Figure 91      Recambering Design C, Second Vane, Tip Defining Section (GG)

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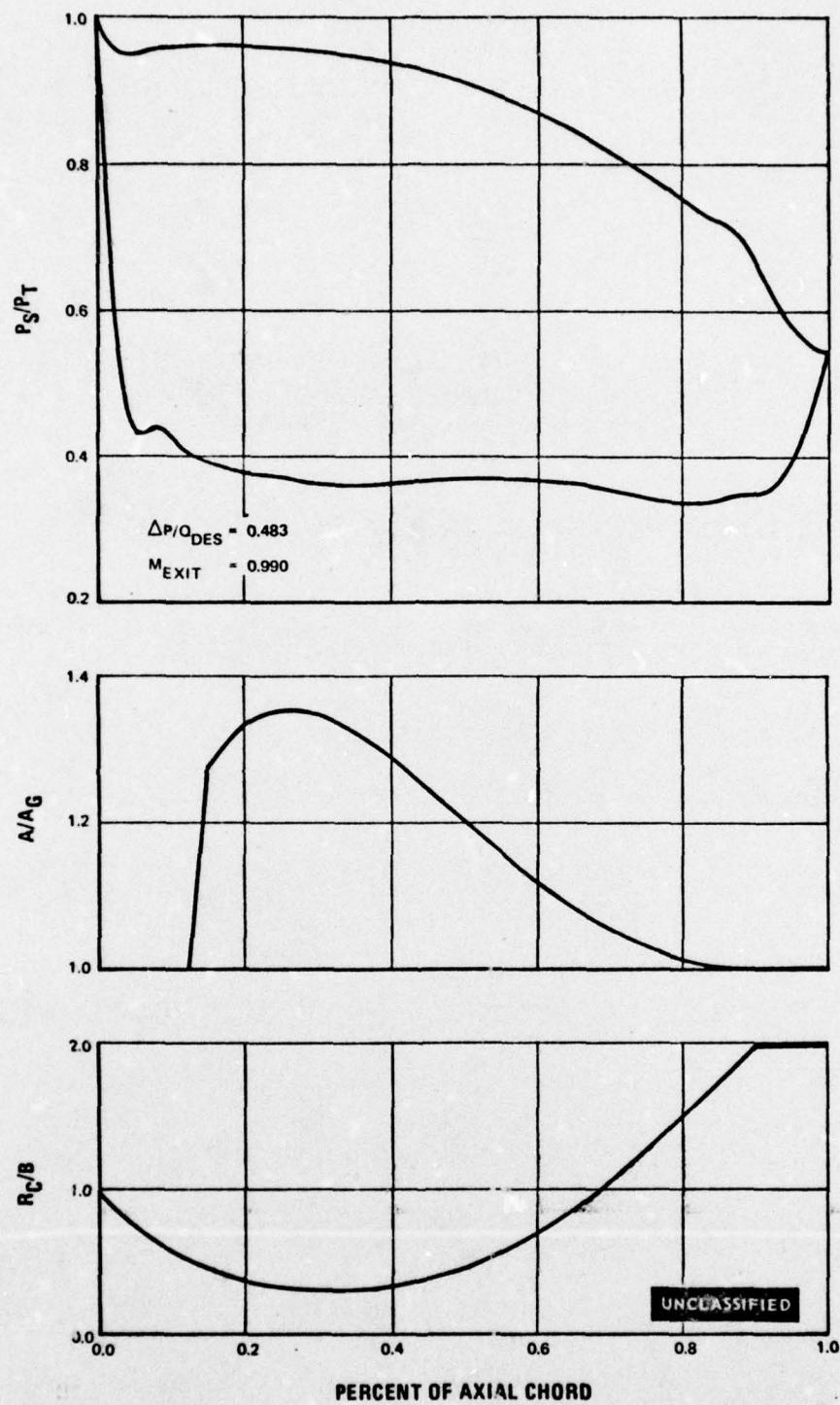
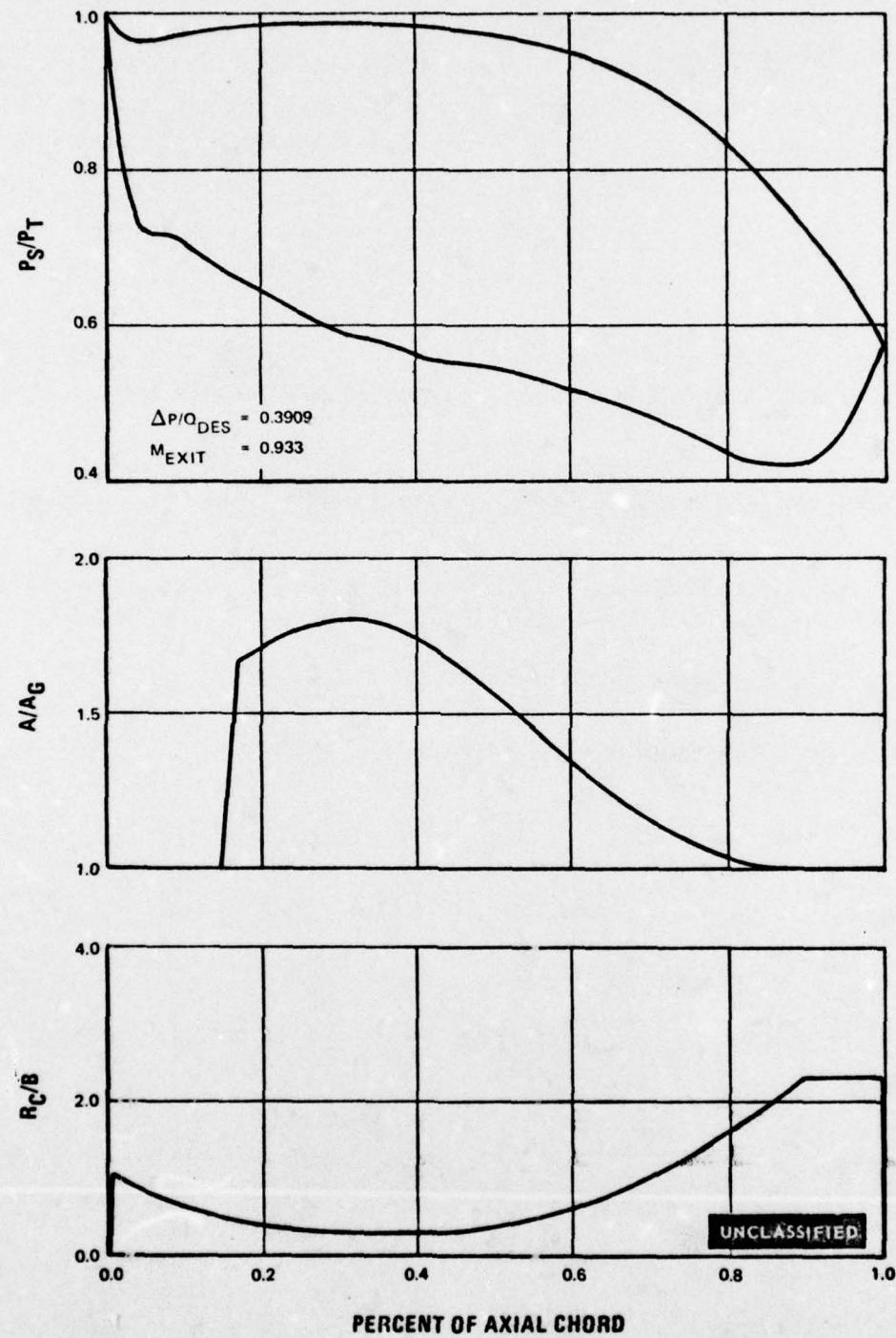


Figure 92 Recambering Design A Second Vane, Root Section

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**Figure 93      Recambering Design A, Second Vane, 1/4 Root Section**

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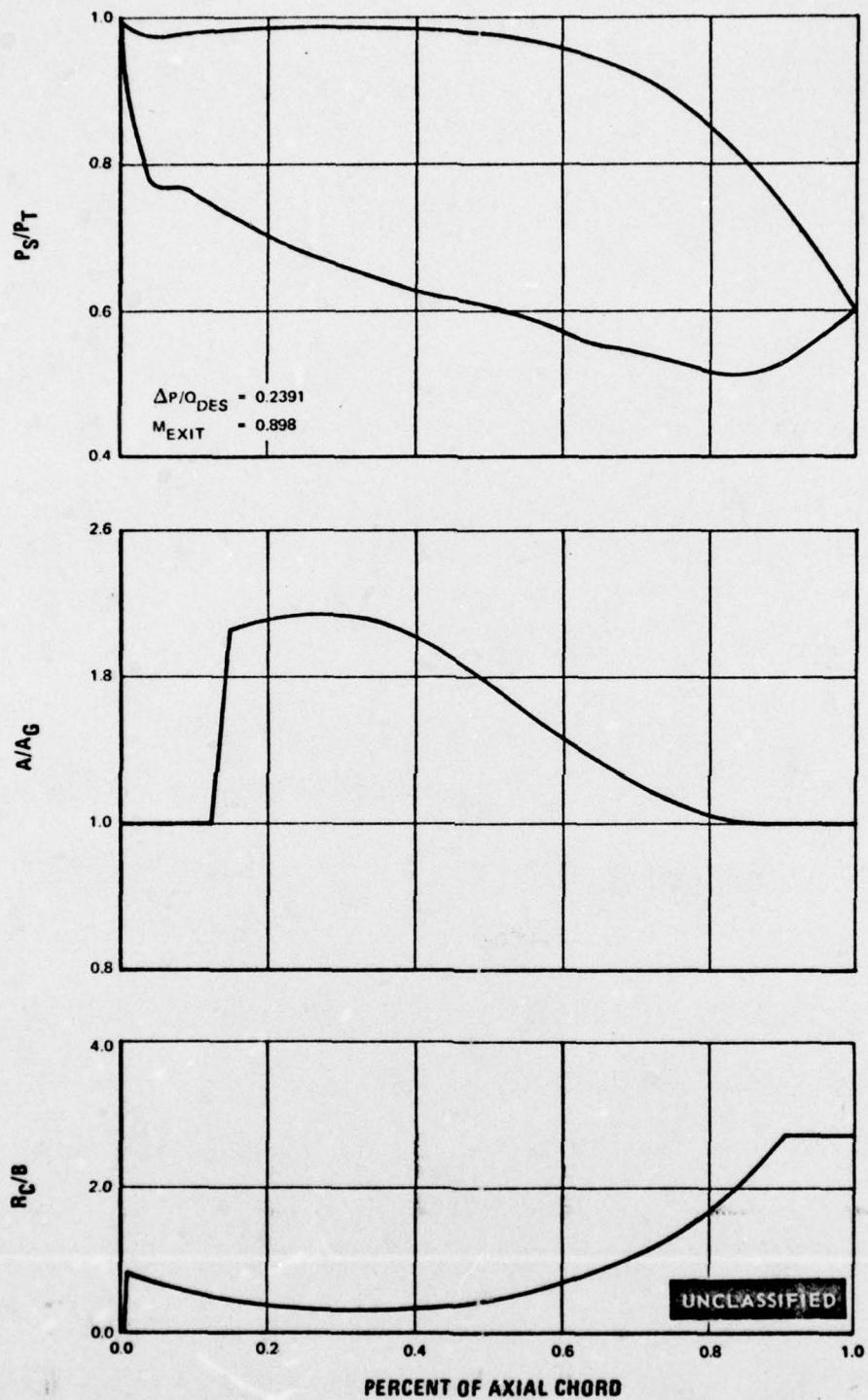


Figure 94 Recamering Design A., Second Vane, Mean Section

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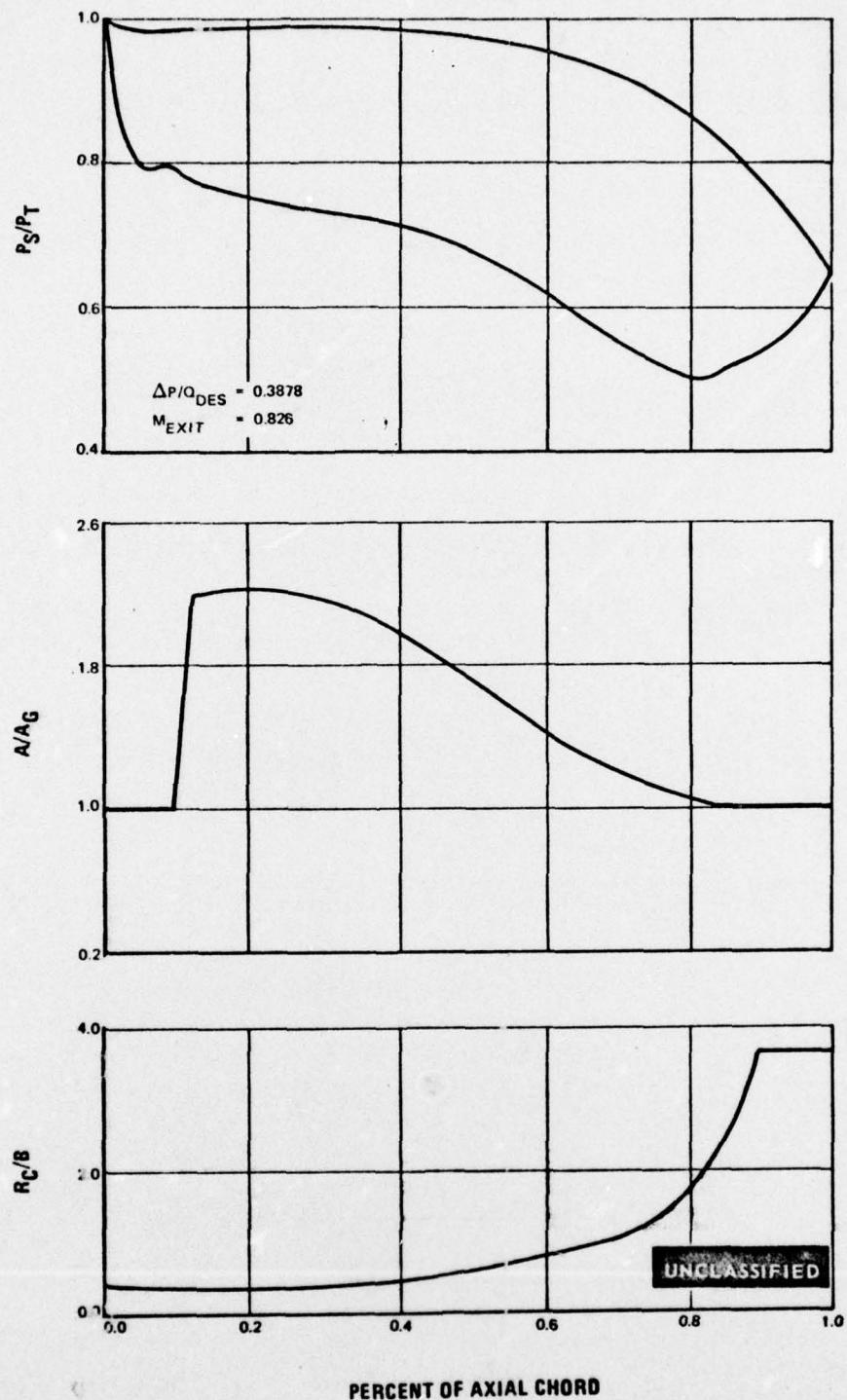


Figure 95      Recamering Design A, Second Vane, 1/4 Tip Section

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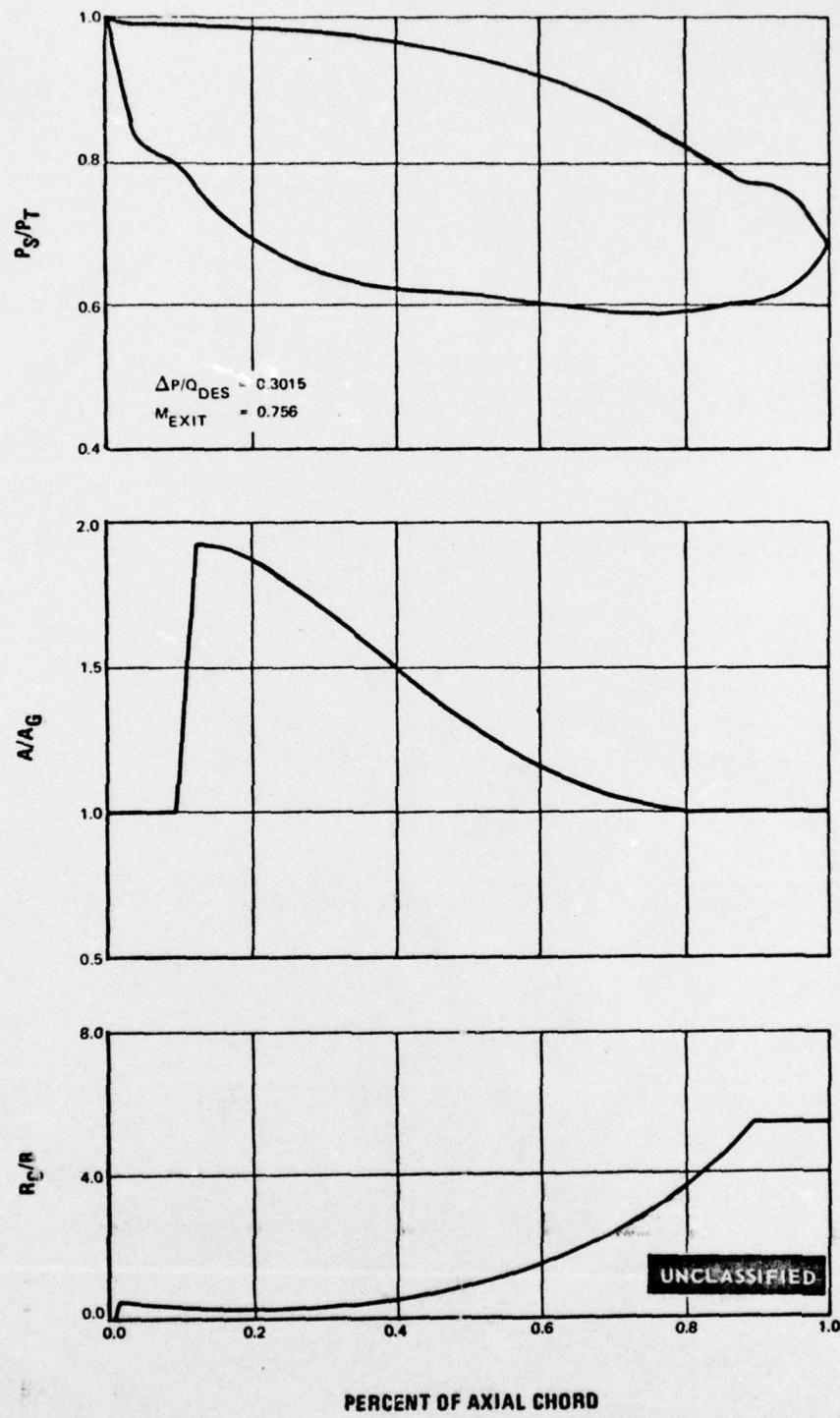


Figure 96 Recamering Design A, Second Vane, Tip Section

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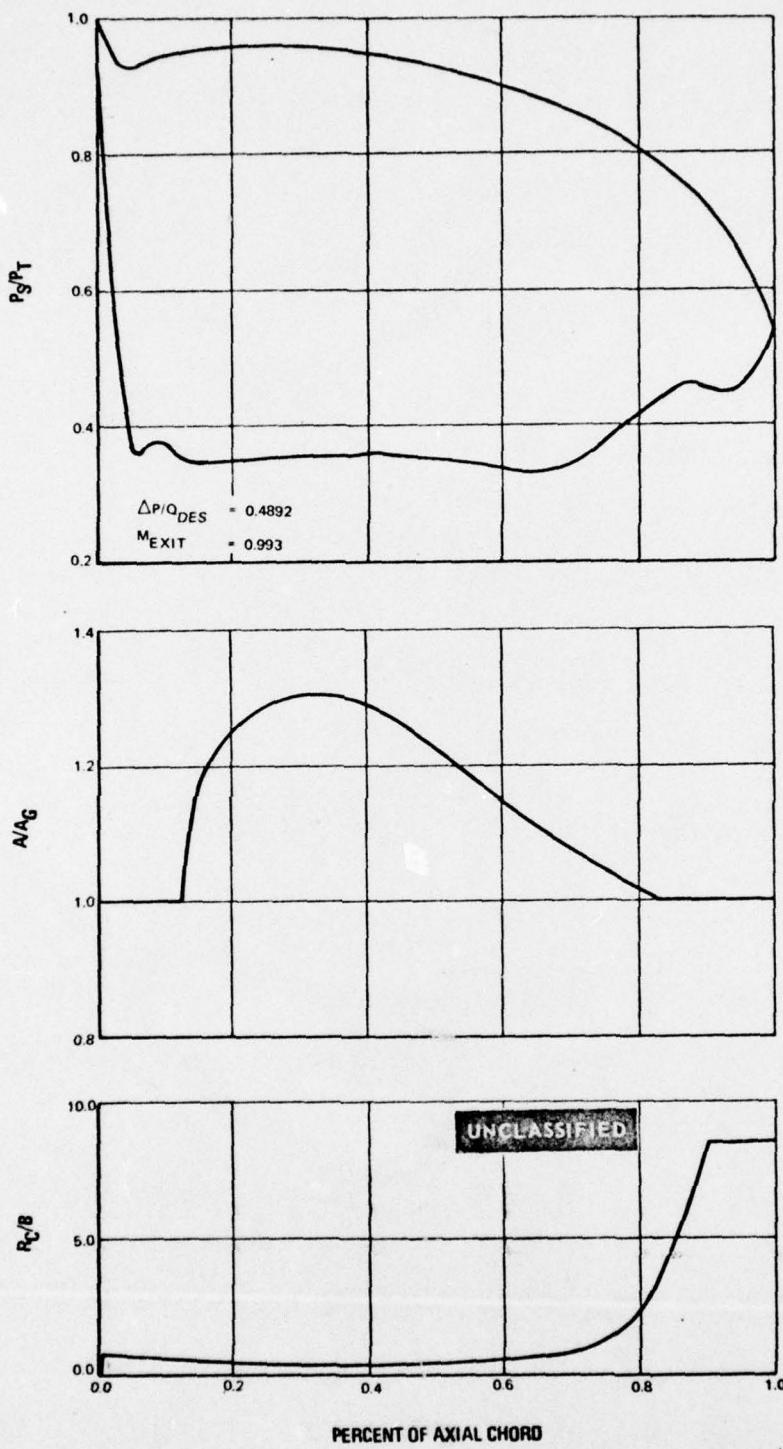


Figure 97 Recambering Design B, Second Vane, Root Section

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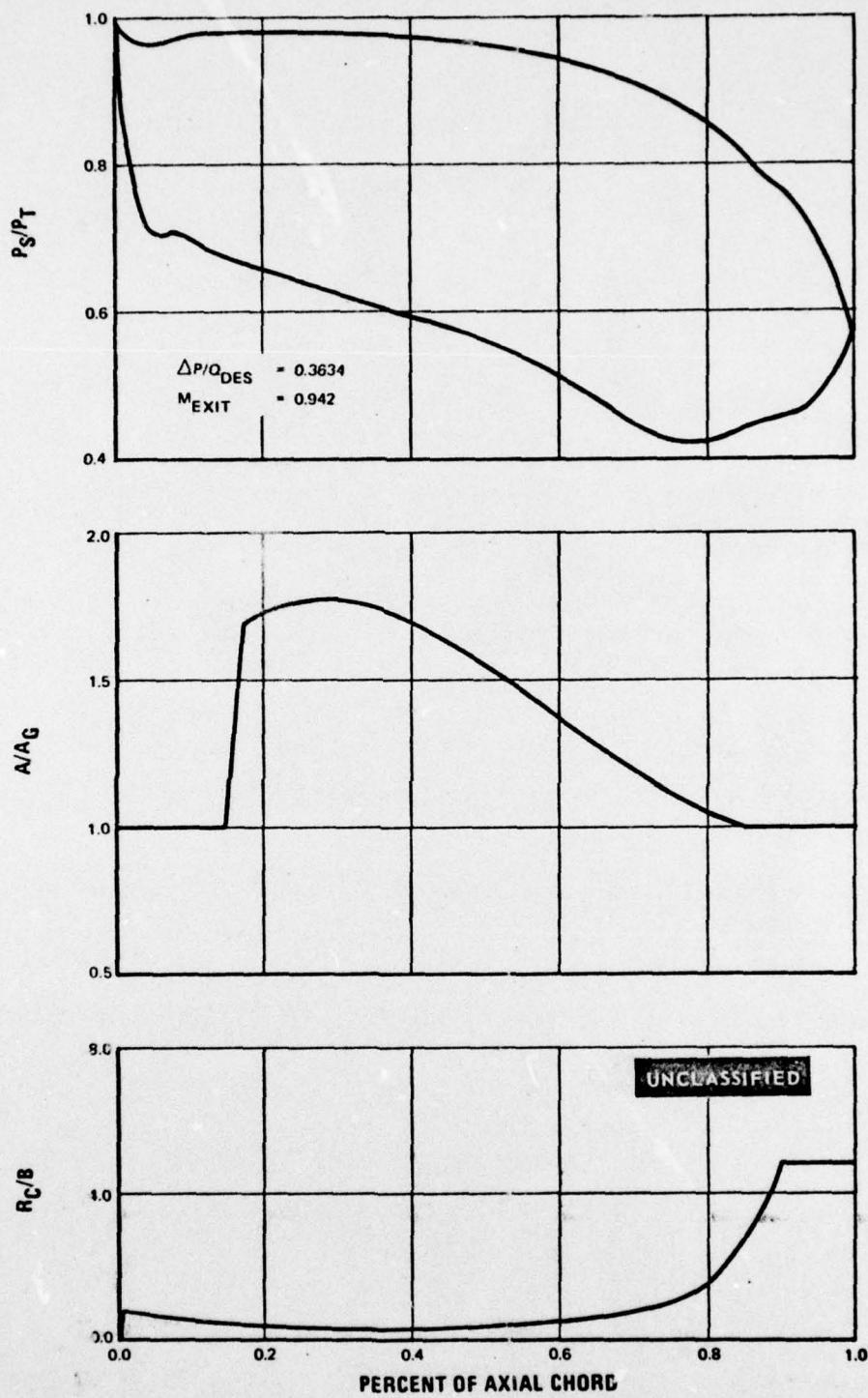


Figure 98 Recambering Design B, Second Vane, 1/4 Root Section

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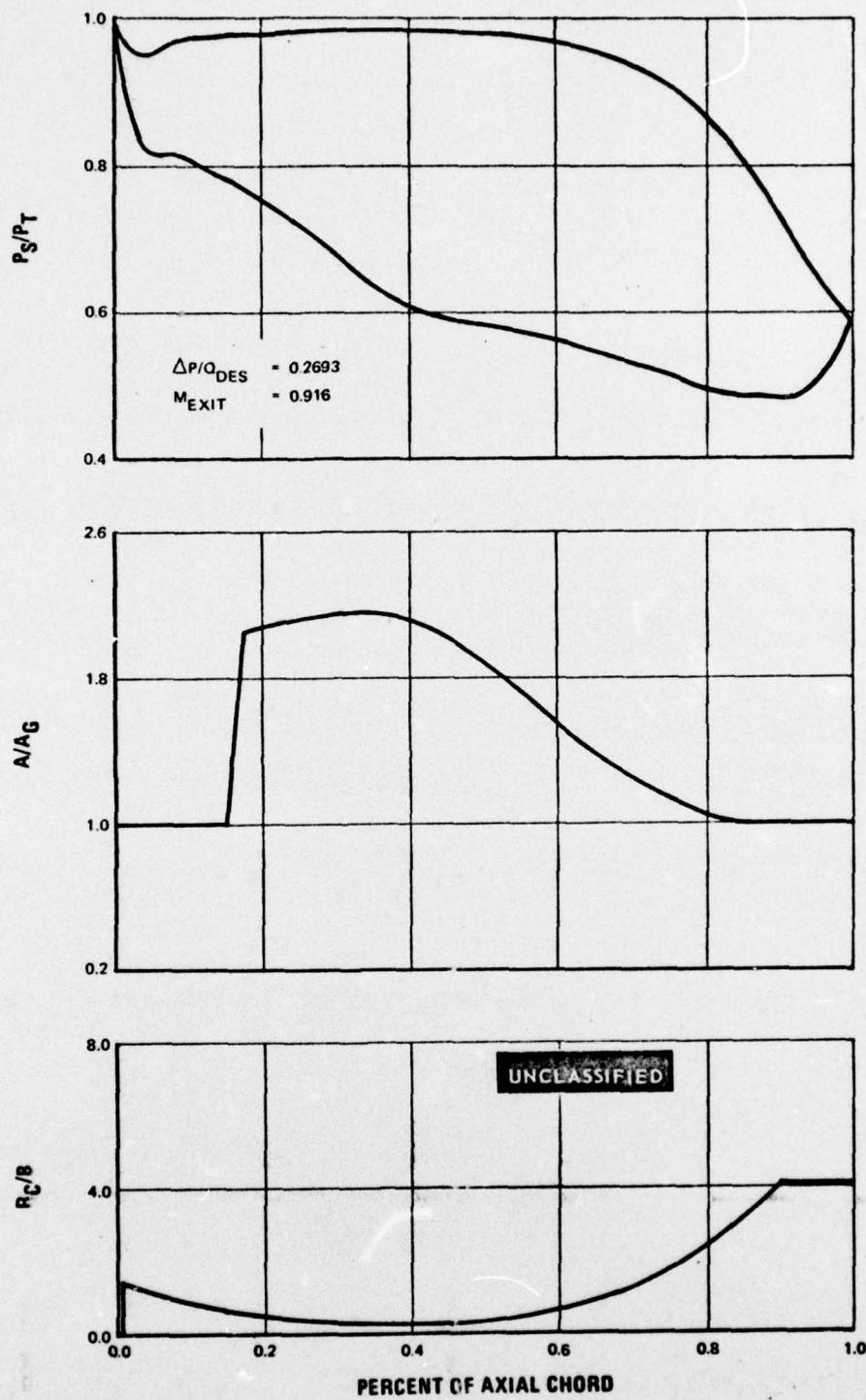


Figure 99 Recambering Design B, Second Vane, Mean Section

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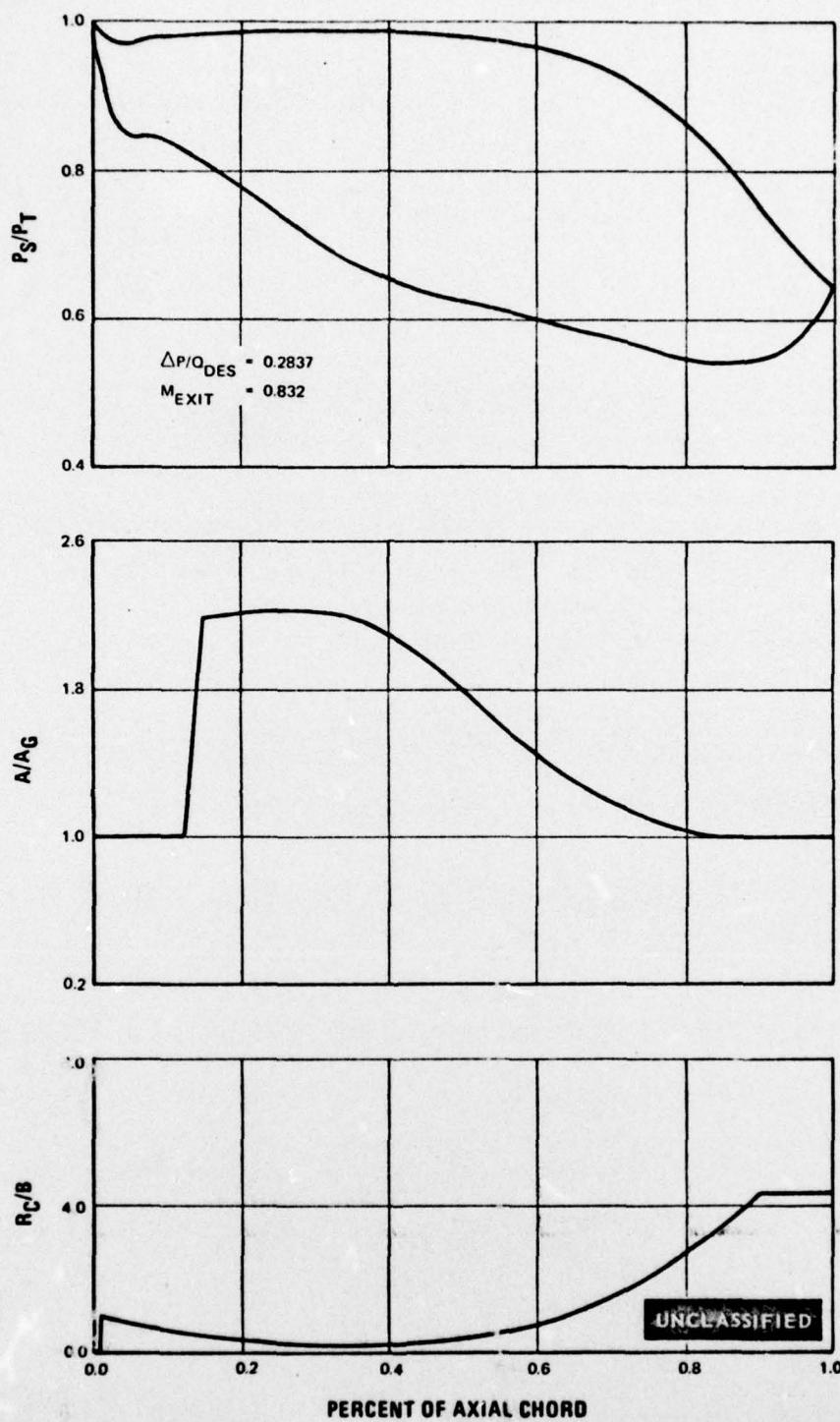


Figure 100 Recamering Design B, Second Vane, 1/4 Tip Section

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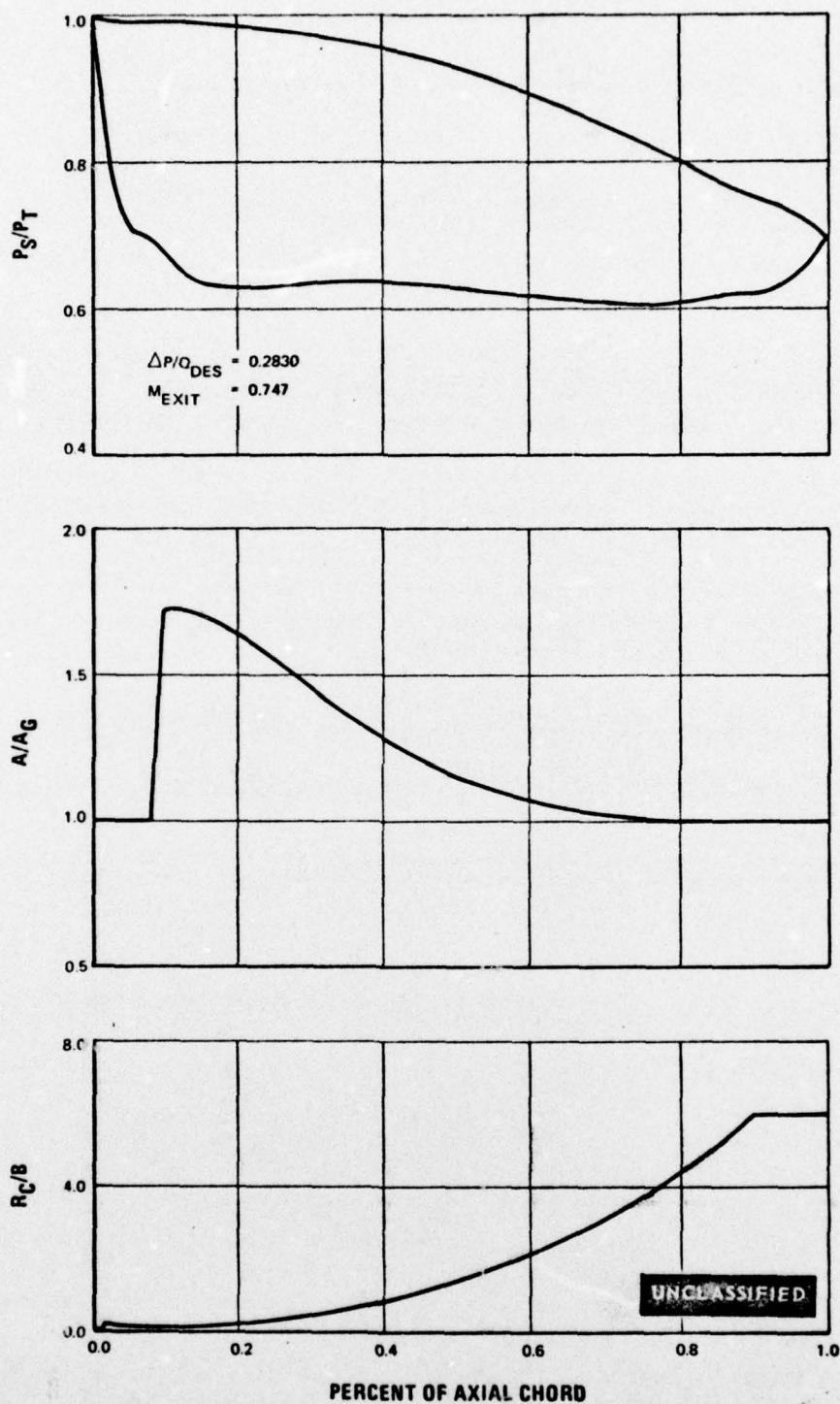


Figure 101 Recamembering Design B, Second Vane, Tip Section

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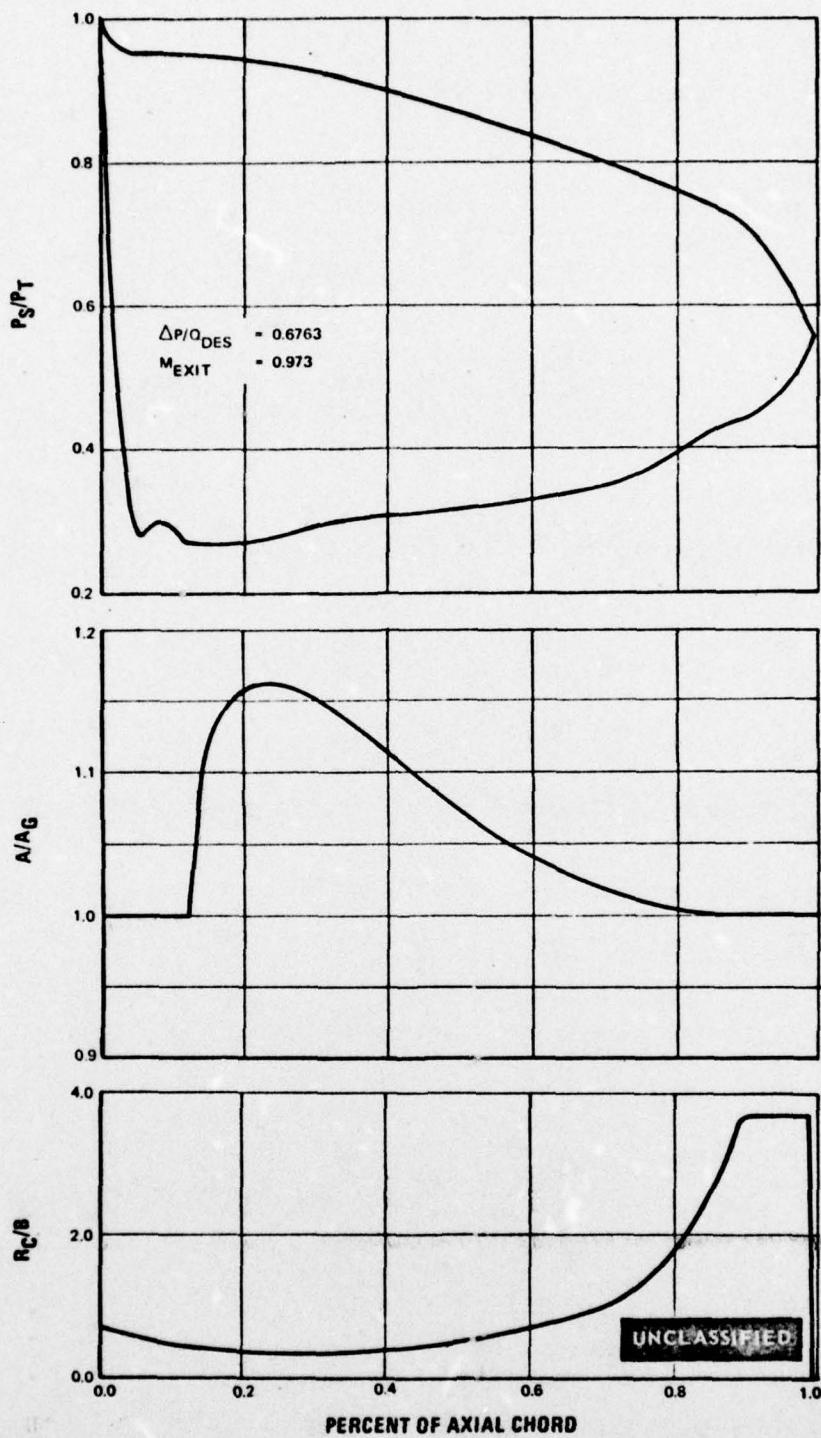


Figure 102 Recambeing Design C, Second Vane, Root Section

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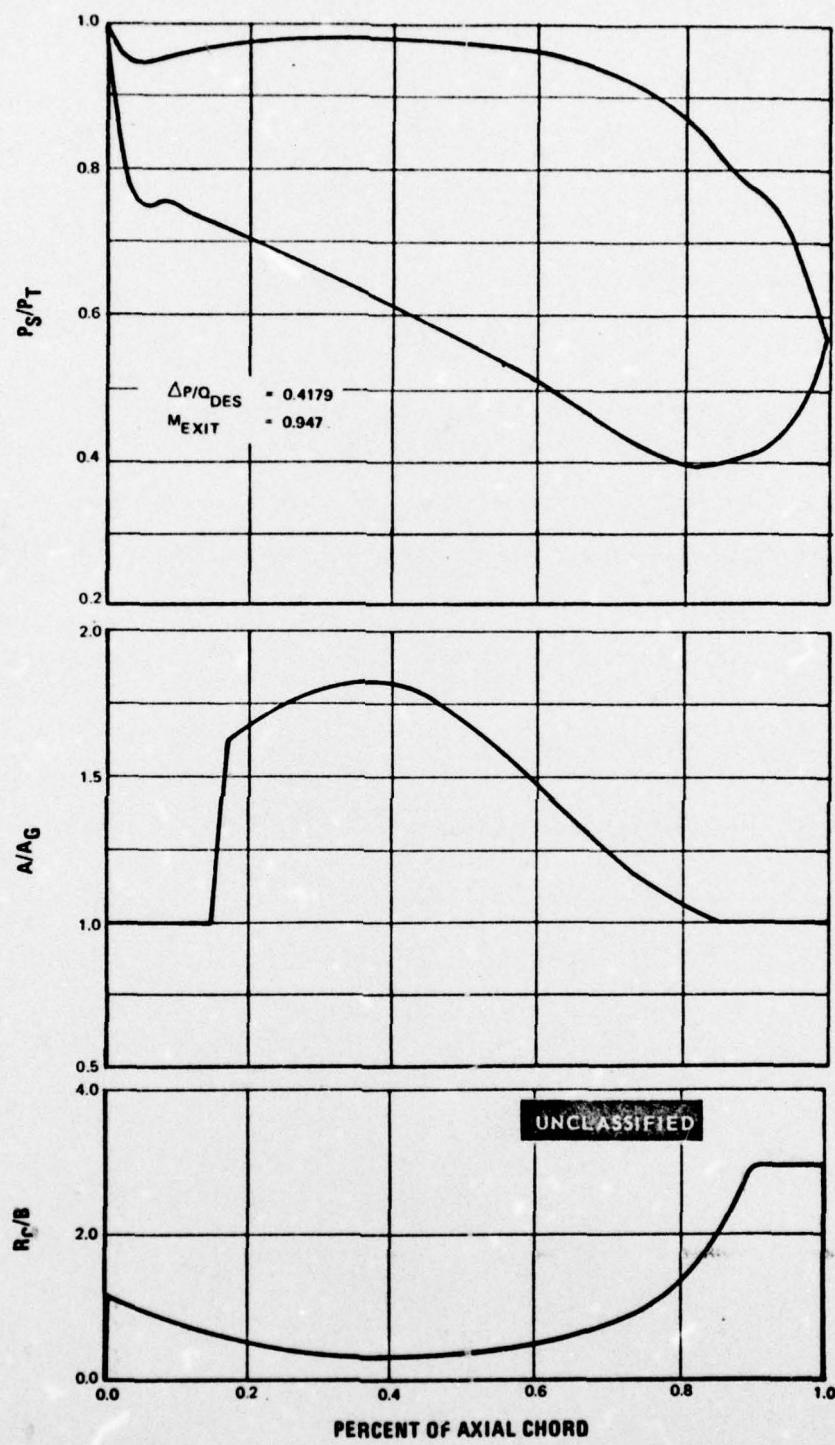


Figure 103 Recamering Design C, Second Vane, 1/4 Root Section

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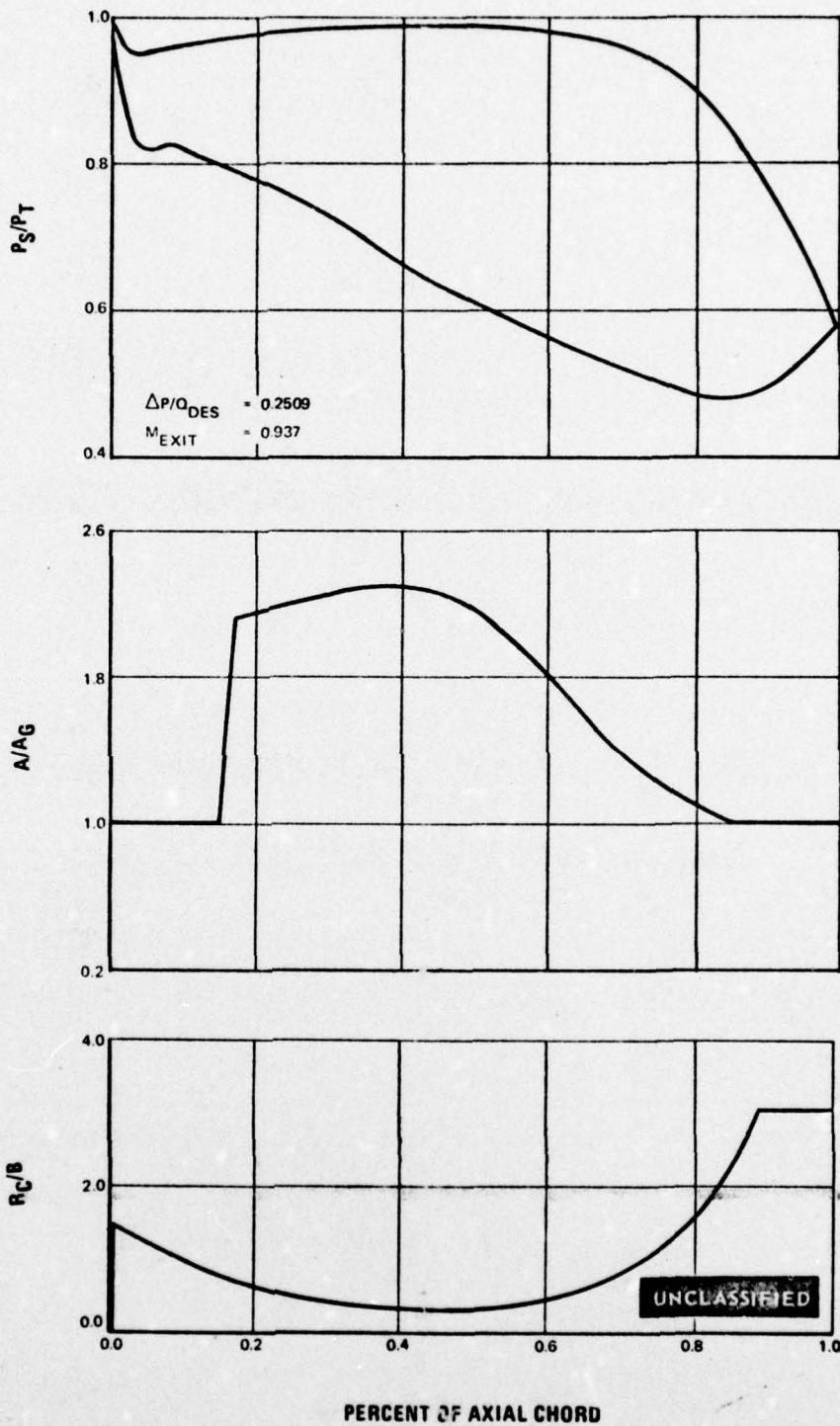


Figure 104 Recambering Design C, Second Vane, Mean Section

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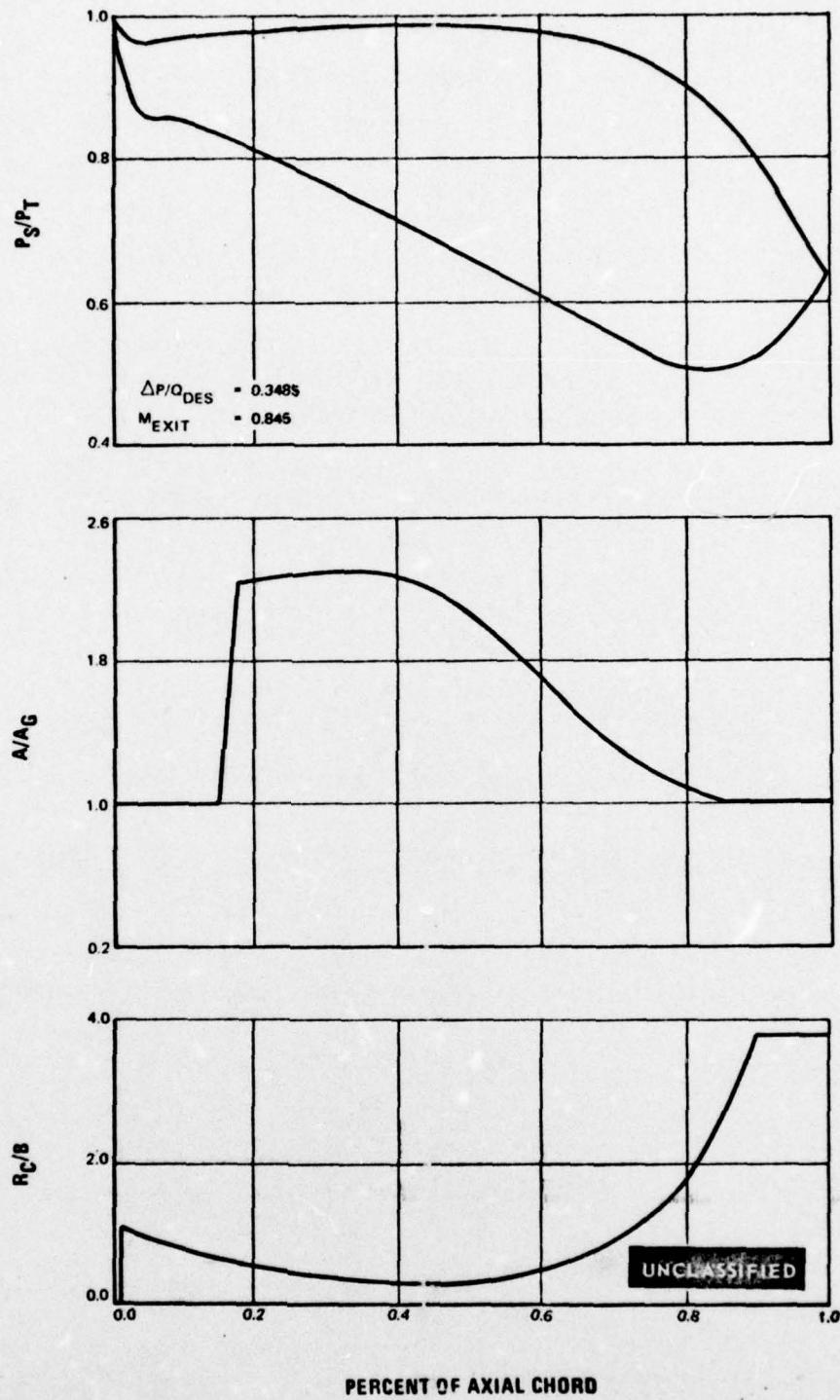


Figure 105 Recamembering Design C, Second Vane, 1/4 Tip Section

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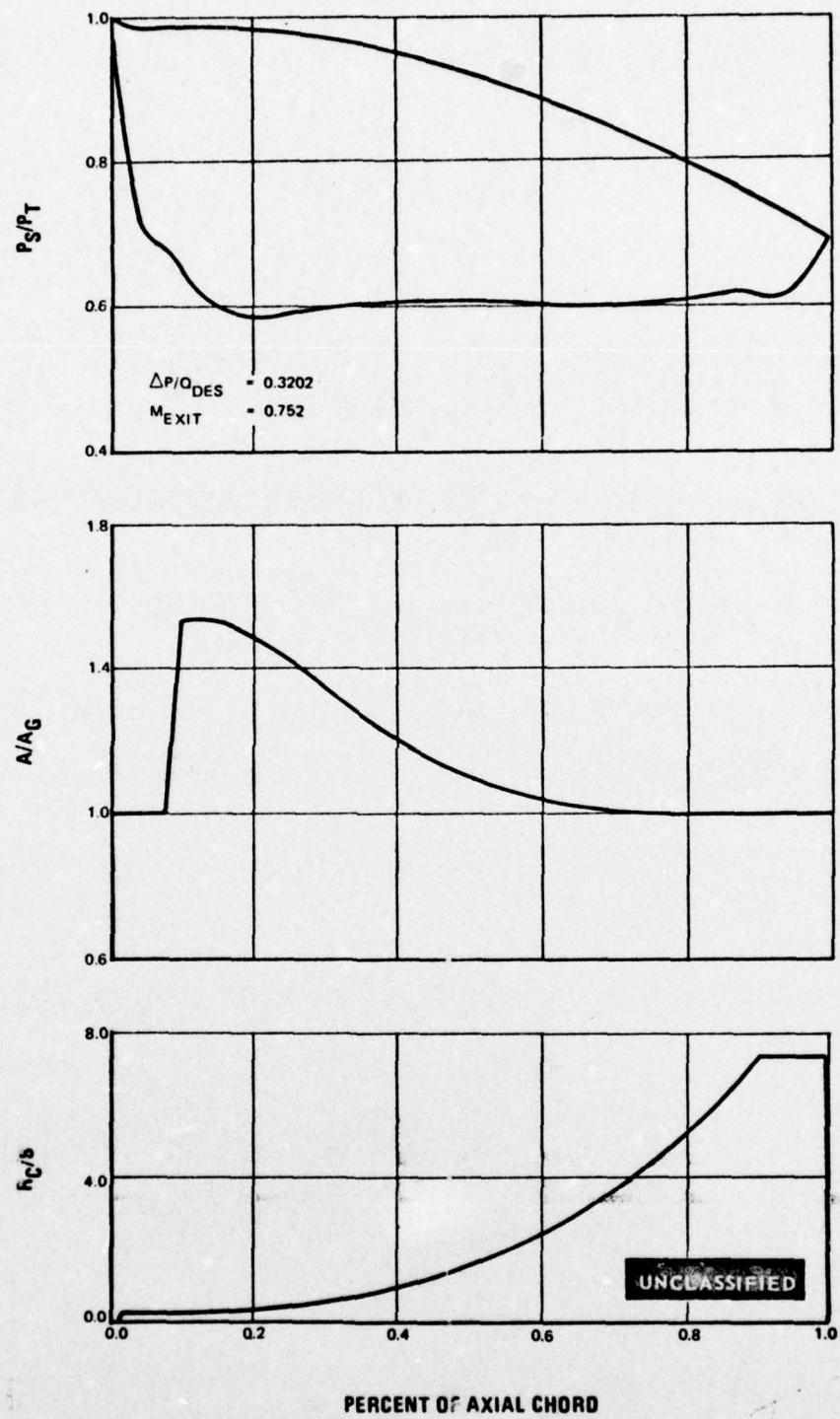


Figure 106 Recambering Design C, Second Vane, Tip Section

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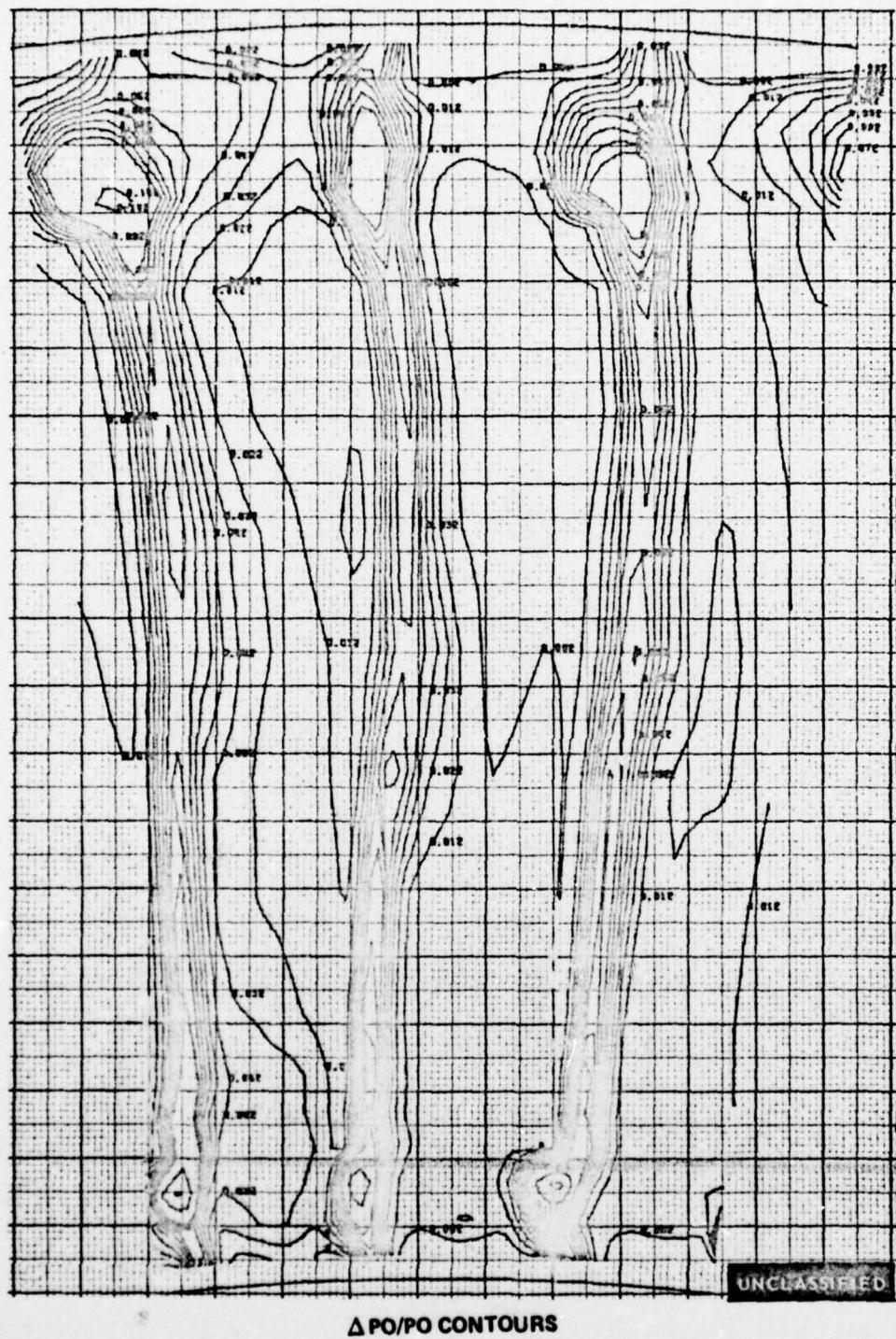
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(U) Except for the test airfoils, the cascade rig used in this Task was identical to the one used to evaluate the first recambered second vane airfoil (Task II c). This was the normal solidity, annular segment, cascade rig described in detail in Reference 2. The inlet guide vanes used in testing the three local recambering designs were the redesigned inlet guide vanes described in Reference 4, Section IV, since the inlet angles of the three recambered airfoils were identical to both the baseline and the first recambered designs. In keeping with the Task II c evaluations, inlet boundary layer bleed slots were also used during these tests. The boundary layer slots were cut into the inlet guide vane pack inside diameter and outside diameter walls, and were 0.094 and 0.047 inches wide, respectively. The inlet screen of Task IIc was also in the cascade in order to generate a more realistic turbulence intensity level.

## **4. DISCUSSION**

(U) The performance data at the exit Mach number nearest the design value (0.869) are presented for each of the three recambering designs. For each geometrical variation, the pressure loss contours, the exit gas angle contours, spanwise pressure loss distributions, spanwise loss coefficient distributions, spanwise exit gas angle distributions and spanwise exit Mach number distributions are shown. These plots are found in Figure 107 through 112 for recambering design A at an exit midspan Mach number of 0.846. Data were also taken at exit midspan Mach numbers of 0.787 and 0.972. Similar plots are found for recambering design B in Figures 113 through 118 at an exit midspan Mach number of 0.856, and for recambering design C at an exit midspan Mach number of 0.862 in Figures 119 through 124. Data were also taken at exit midspan Mach numbers of 0.799 and 0.995 for design B, and at 0.800 and 1.008 for design C.

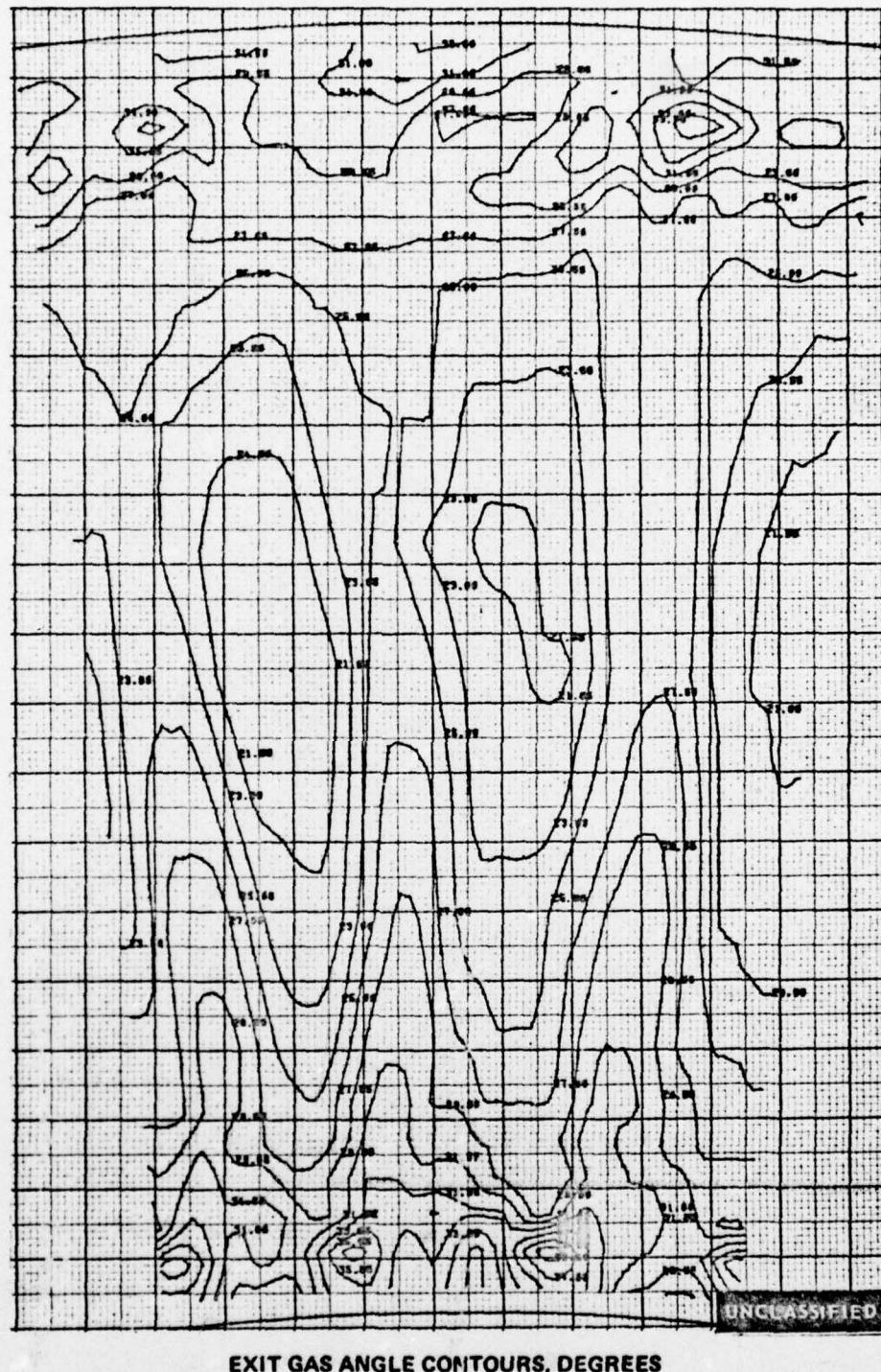
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**Figure 107 Pressure Loss Contours, Second Vane Recambering Design A at Design Incidence. Three Flow Passages. Midspan Exit Mach Number = 0.846**

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**Figure 108** Exit Gas Angle Contours, Second Vane Recambering Design A at Design Incidence. Three Flow Passages. Midspan Exit Mach Number = 0.846

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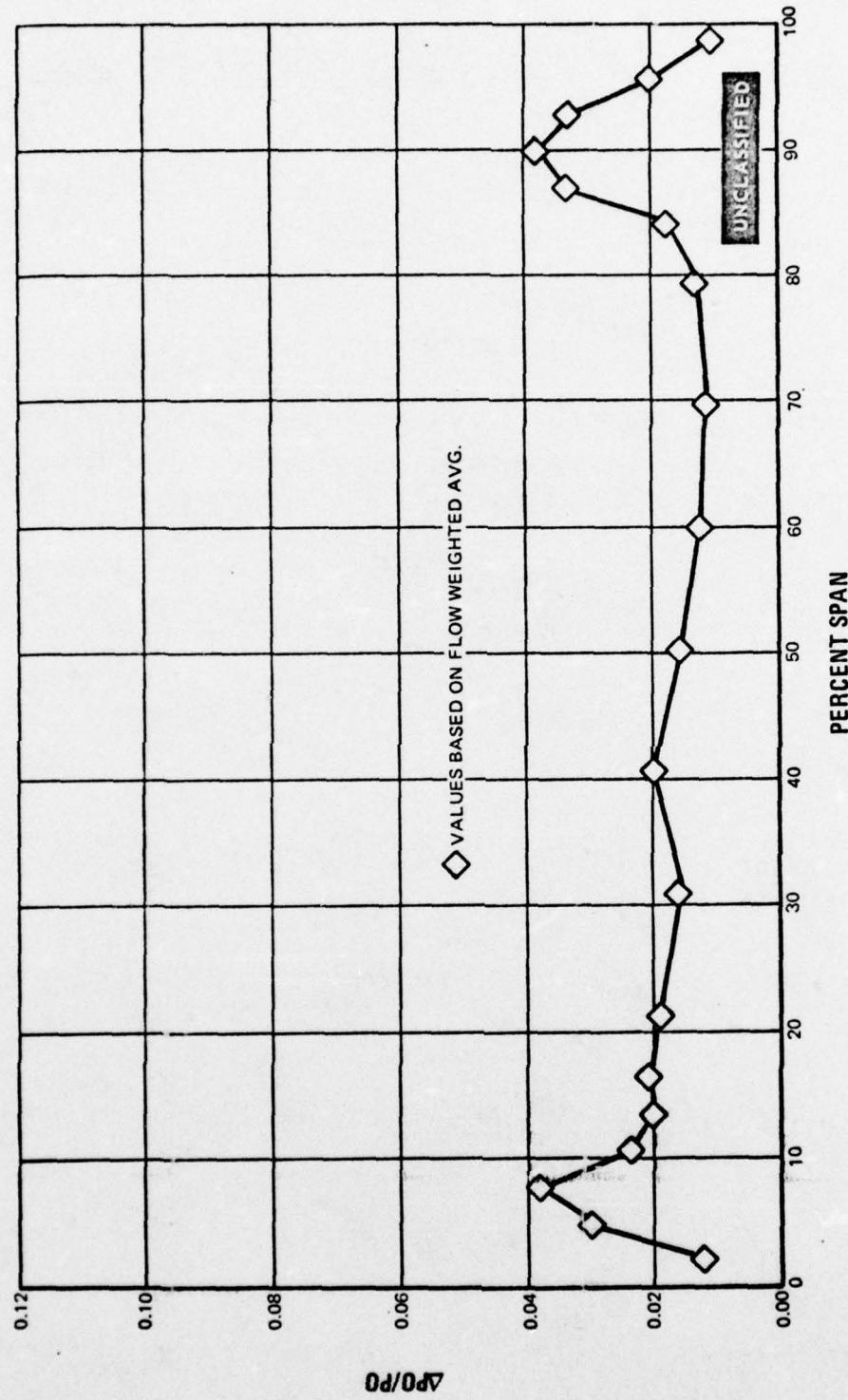


Figure 109 Spanwise Pressure Loss Distribution, Second Vane Recambering Design A  
at Design Incidence. Midspan Exit Mach Number = 0.846

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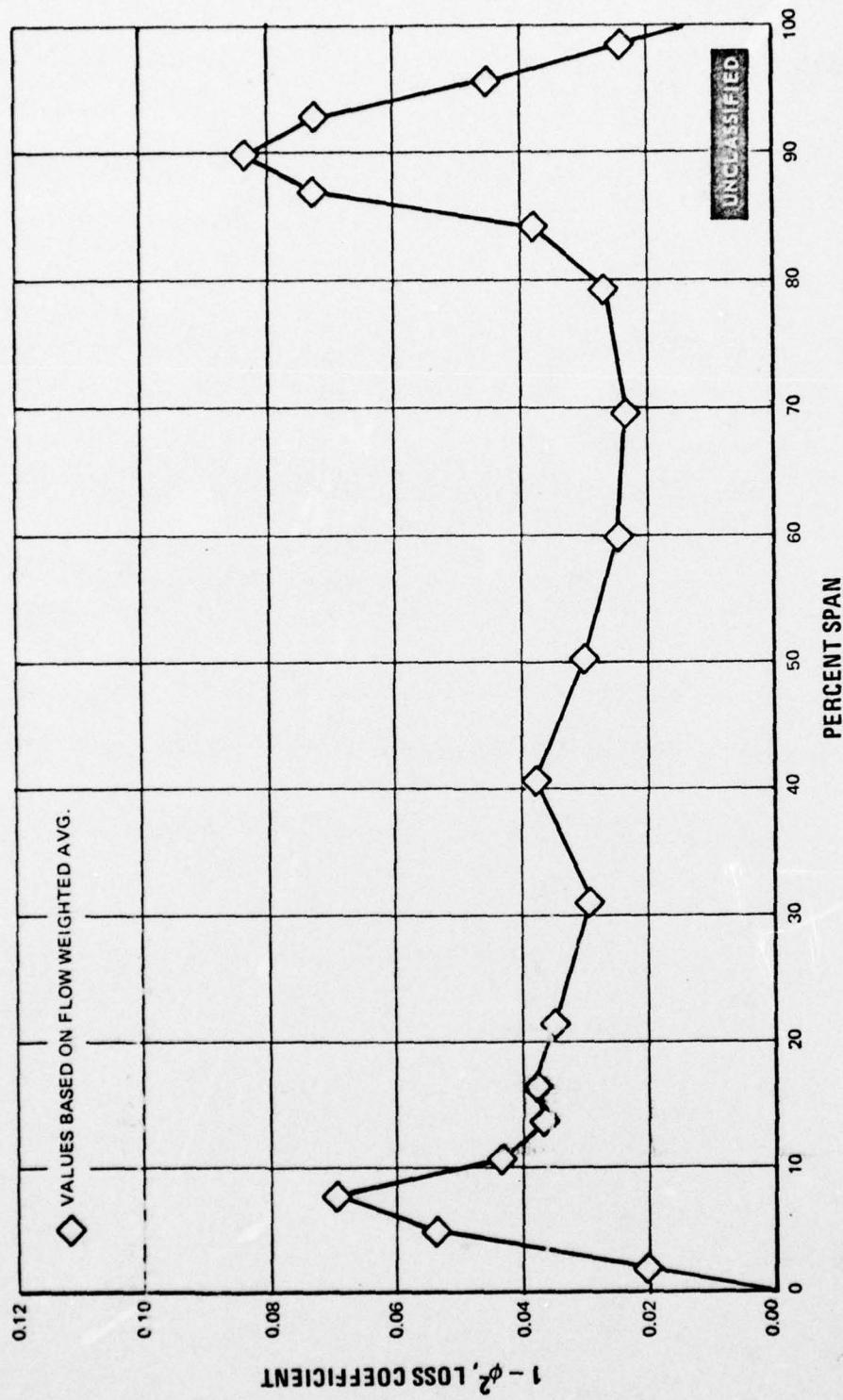


Figure 110 Spanwise Loss Coefficient Distribution, Second Vane Recambering Design A  
at Design Incidence. Midspan Exit Mach Number = 0.846

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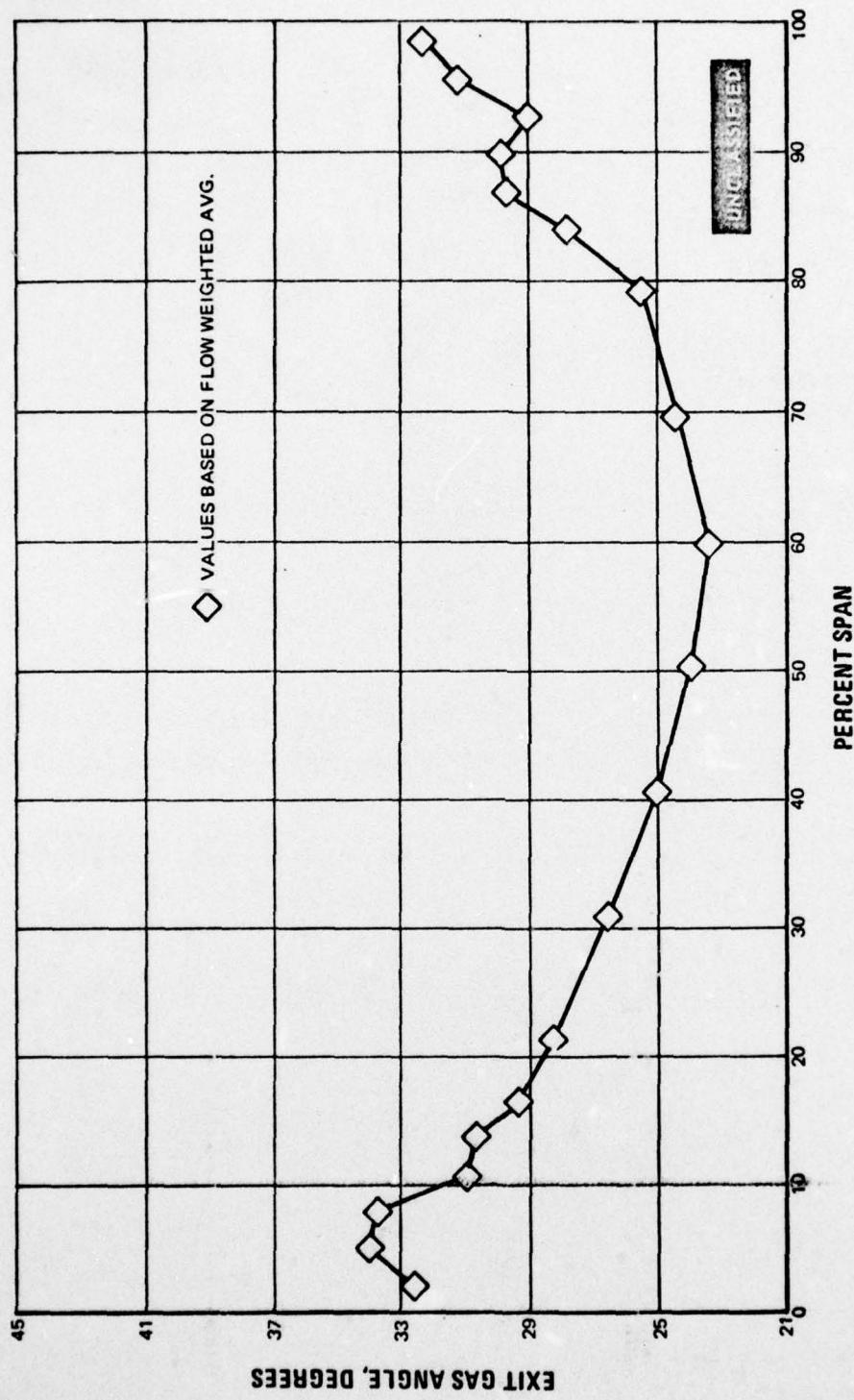


Figure 111 Spanwise Exit Gas Angle Distribution, Second Vane Recambering Design A  
at Design Incidence. Midspan Exit Mach Number = 0.846

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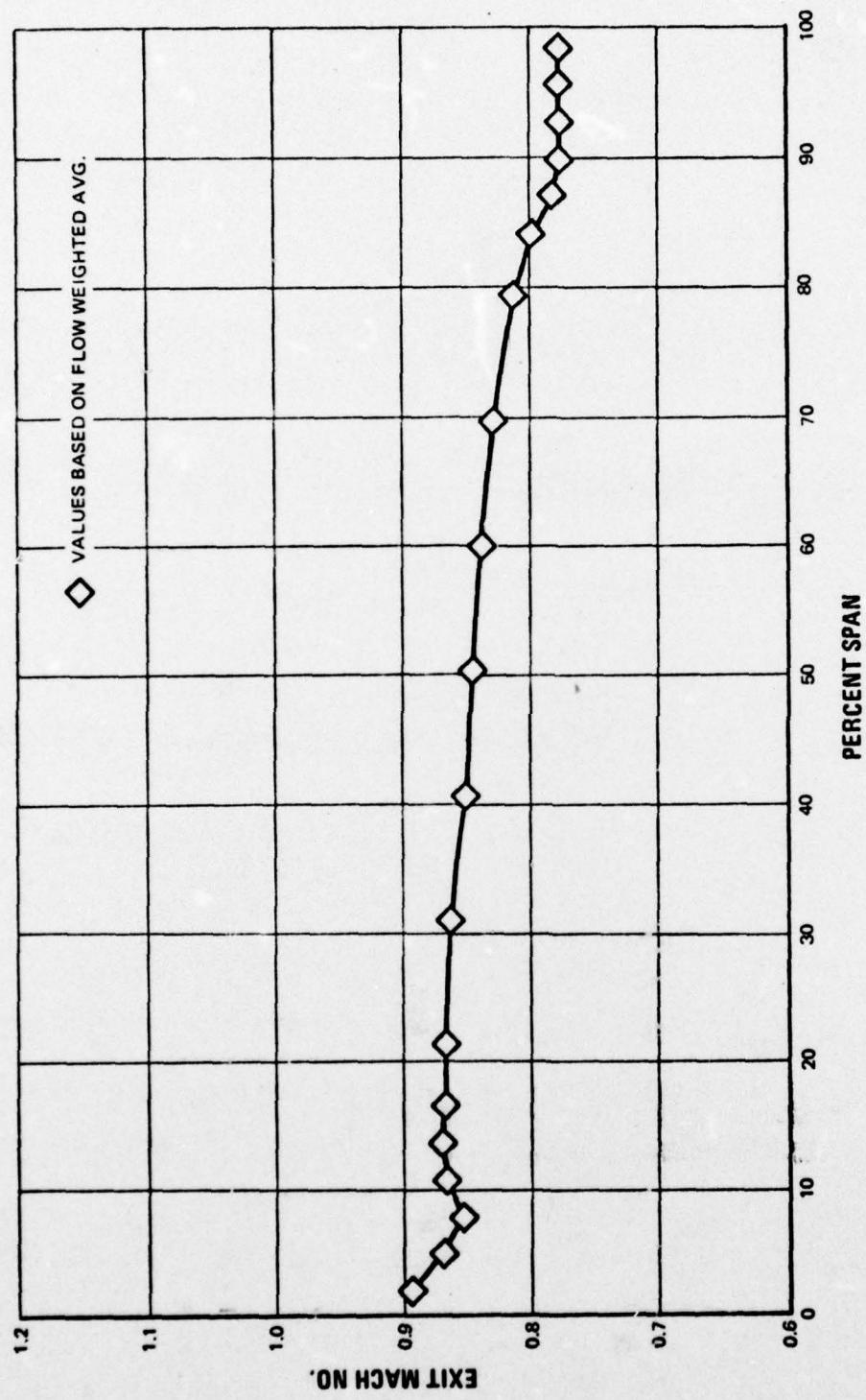


Figure 112 Spanwise Exit Mach Number Distribution, Second Vane Recambering Design  
A at Design Incidence. Midspan Exit Mach Number = 0.846

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Figure 113 Pressure Loss Contours, Second Vane Recambering Design B at Design Incidence. Three Flow Passages. Midspan Exit Mach Number = 0.856

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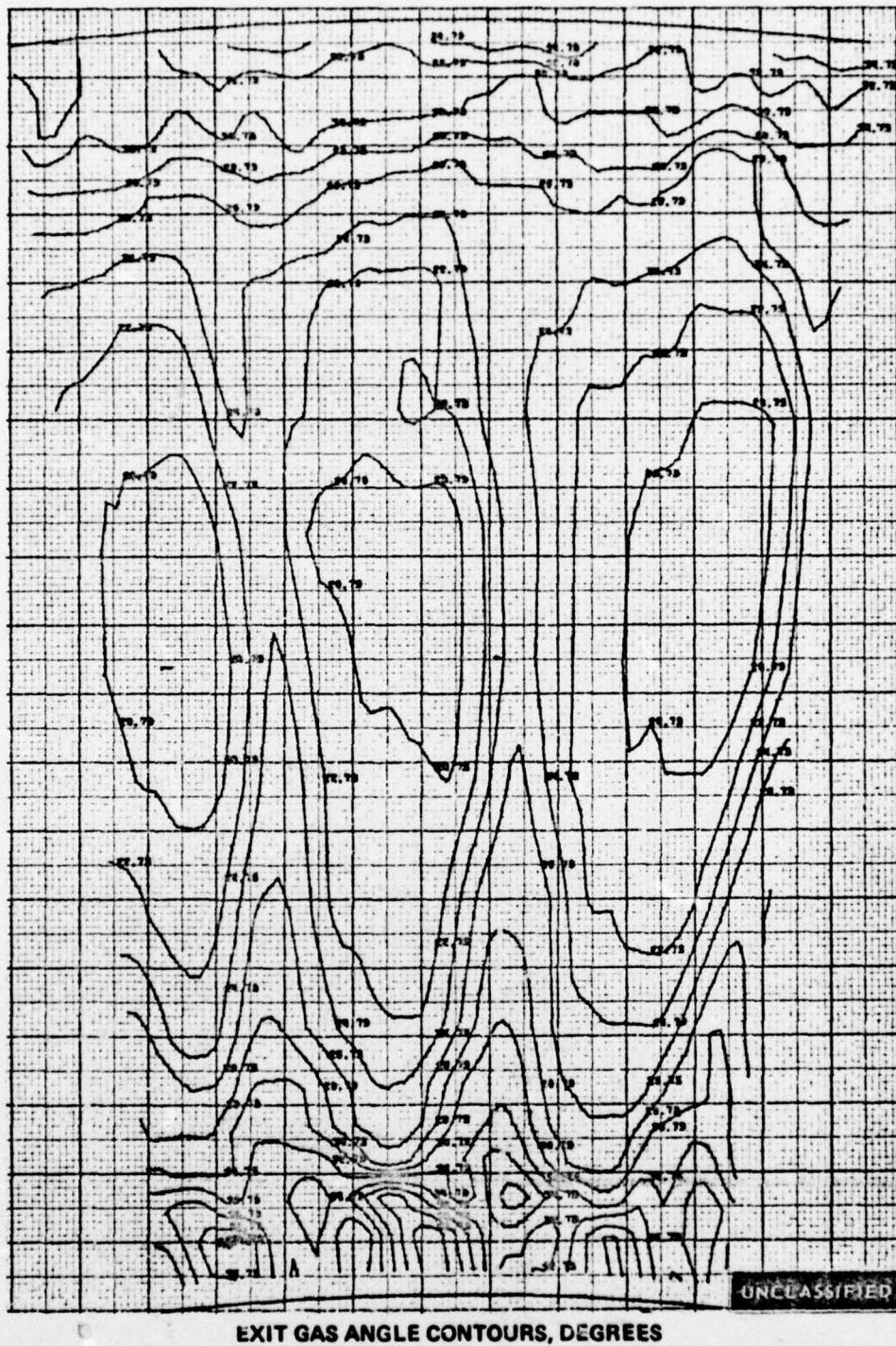


Figure 114 Exit Gas Angle Contours, Second Vane Recambering Design B at Design Incidence. Three Flow Passages. Midspan Exit Mach Number = 0.856

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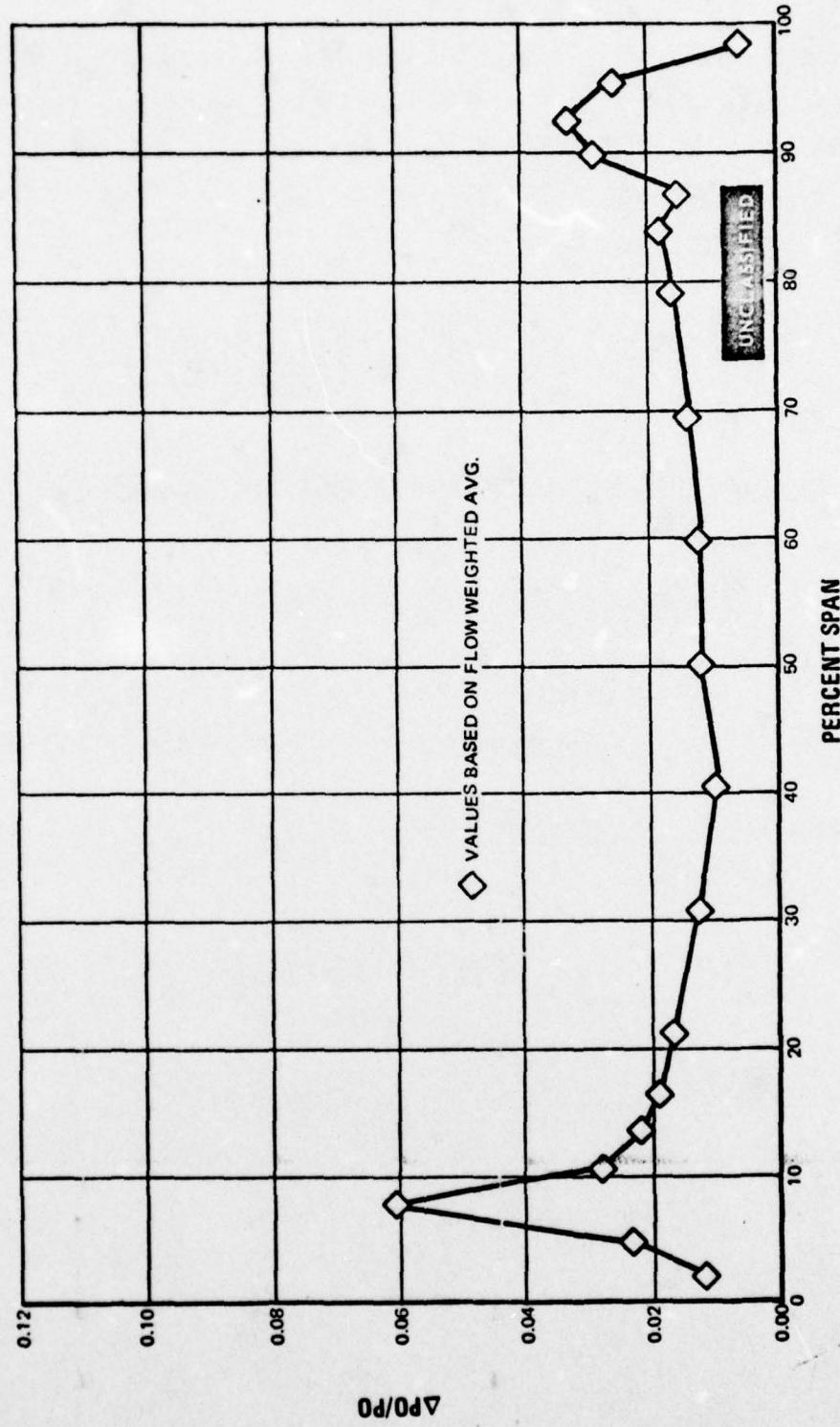


Figure 115 Spanwise Pressure Loss Distribution, Second Vane Recambering Design B  
at Design Incidence. Midspan Exit Mach Number = 0.856

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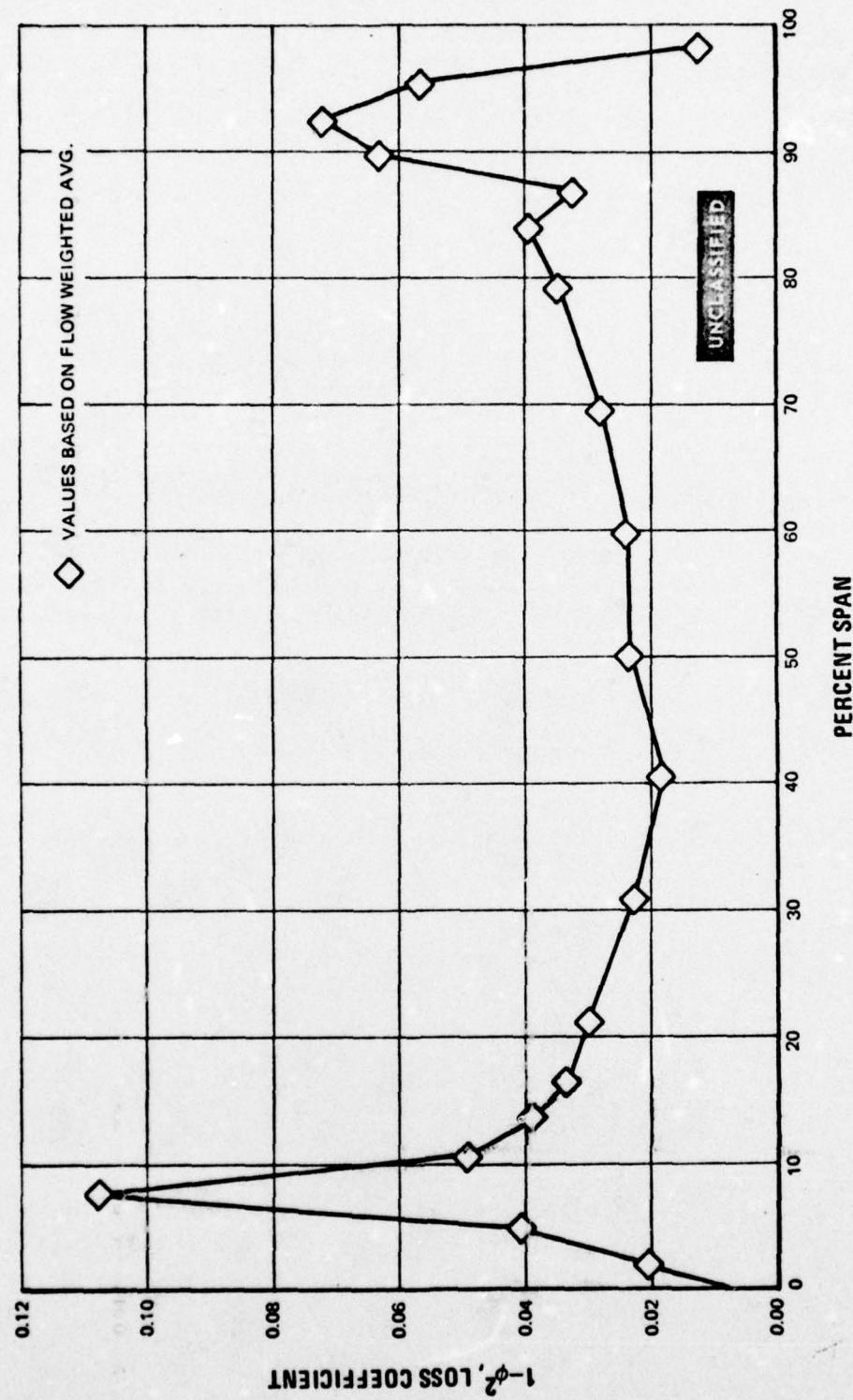


Figure 116 Spanwise Loss Coefficient Distribution, Second Vane Recambering Design B  
at Design Incidence. Midspan Exit Mach Number = 0.856

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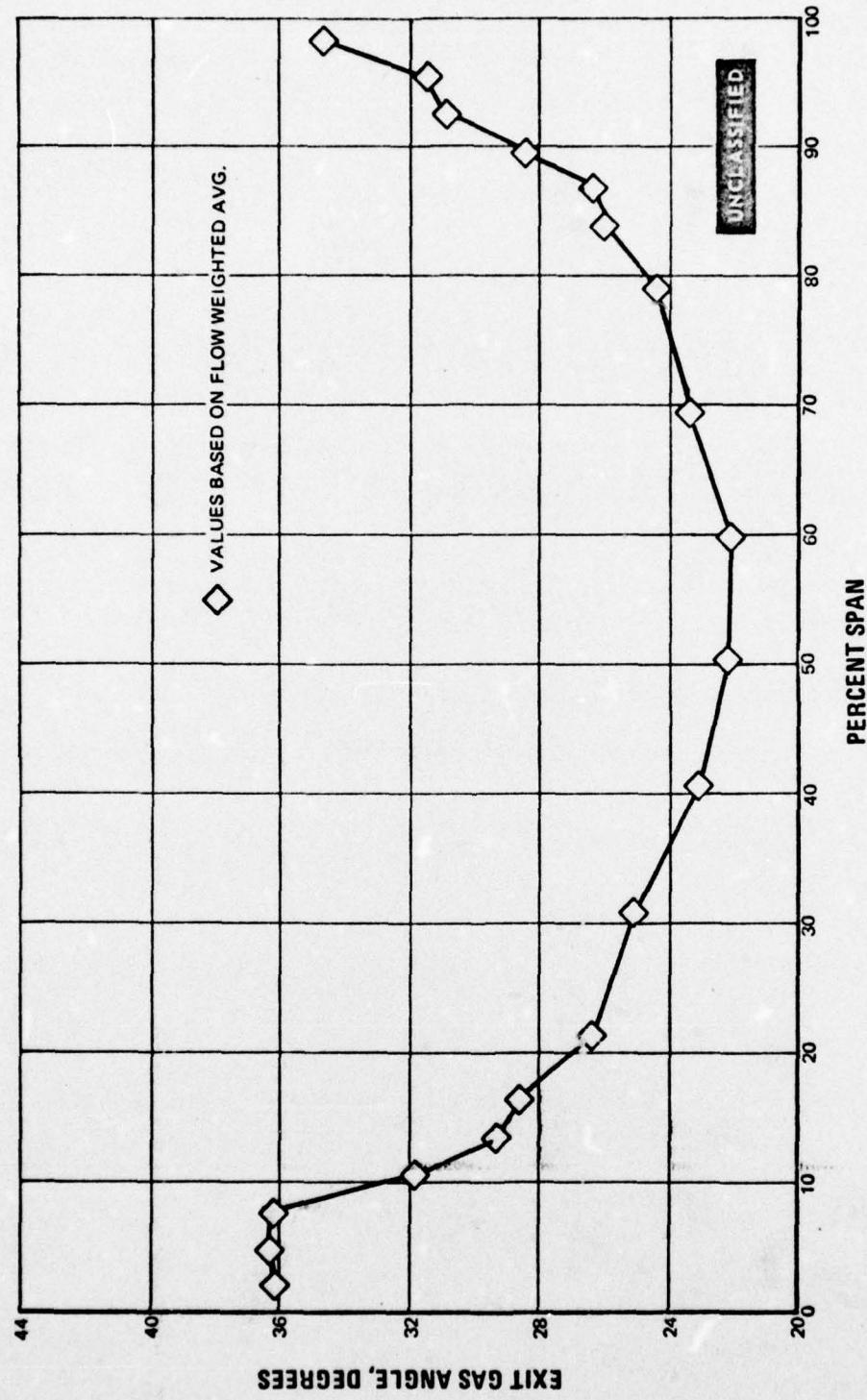


Figure 117 Spanwise Exit Gas Angle Distribution, Second Vane Recambering Design B  
at Design Incidence. Midspan Exit Mach Number = 0.856

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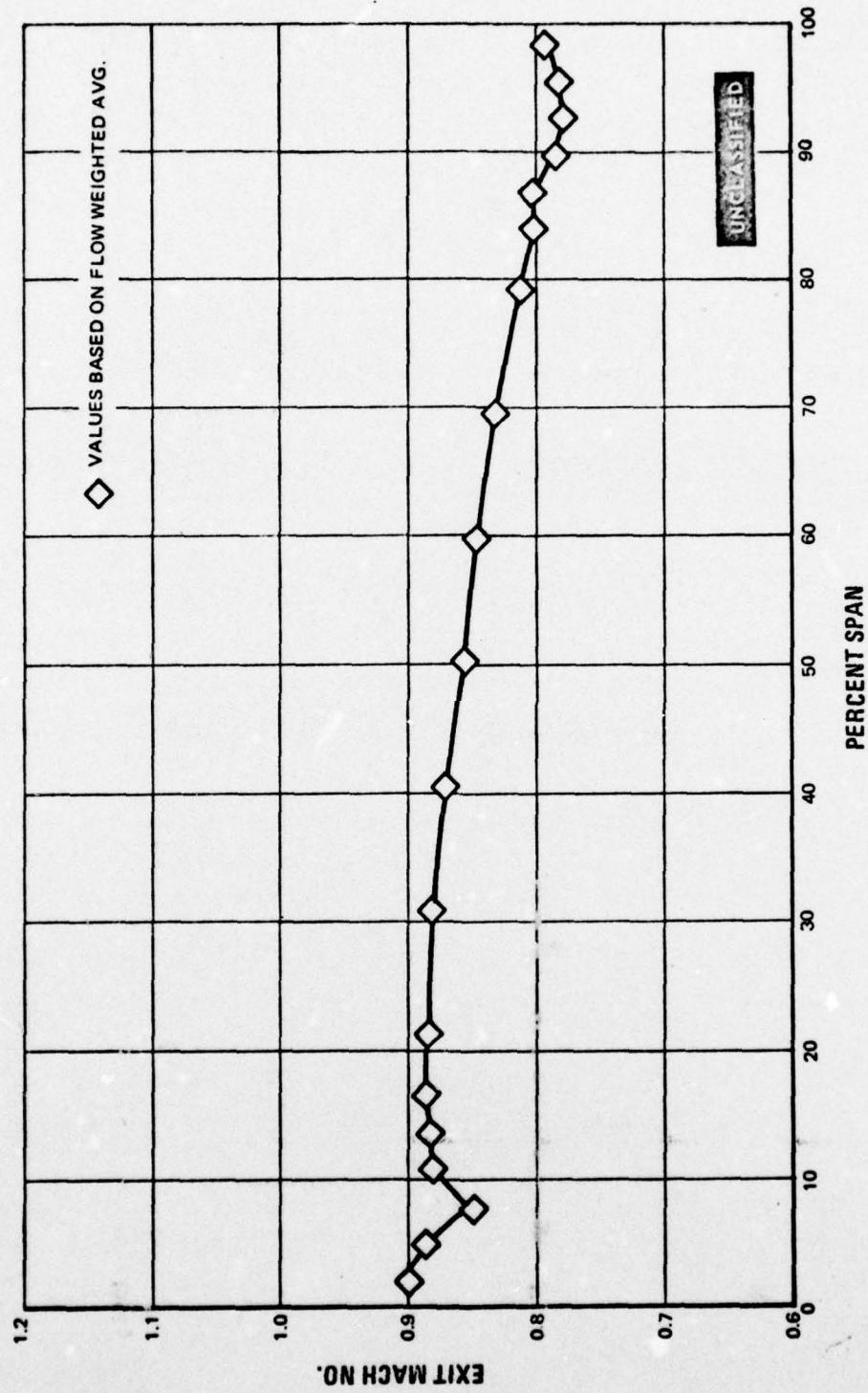
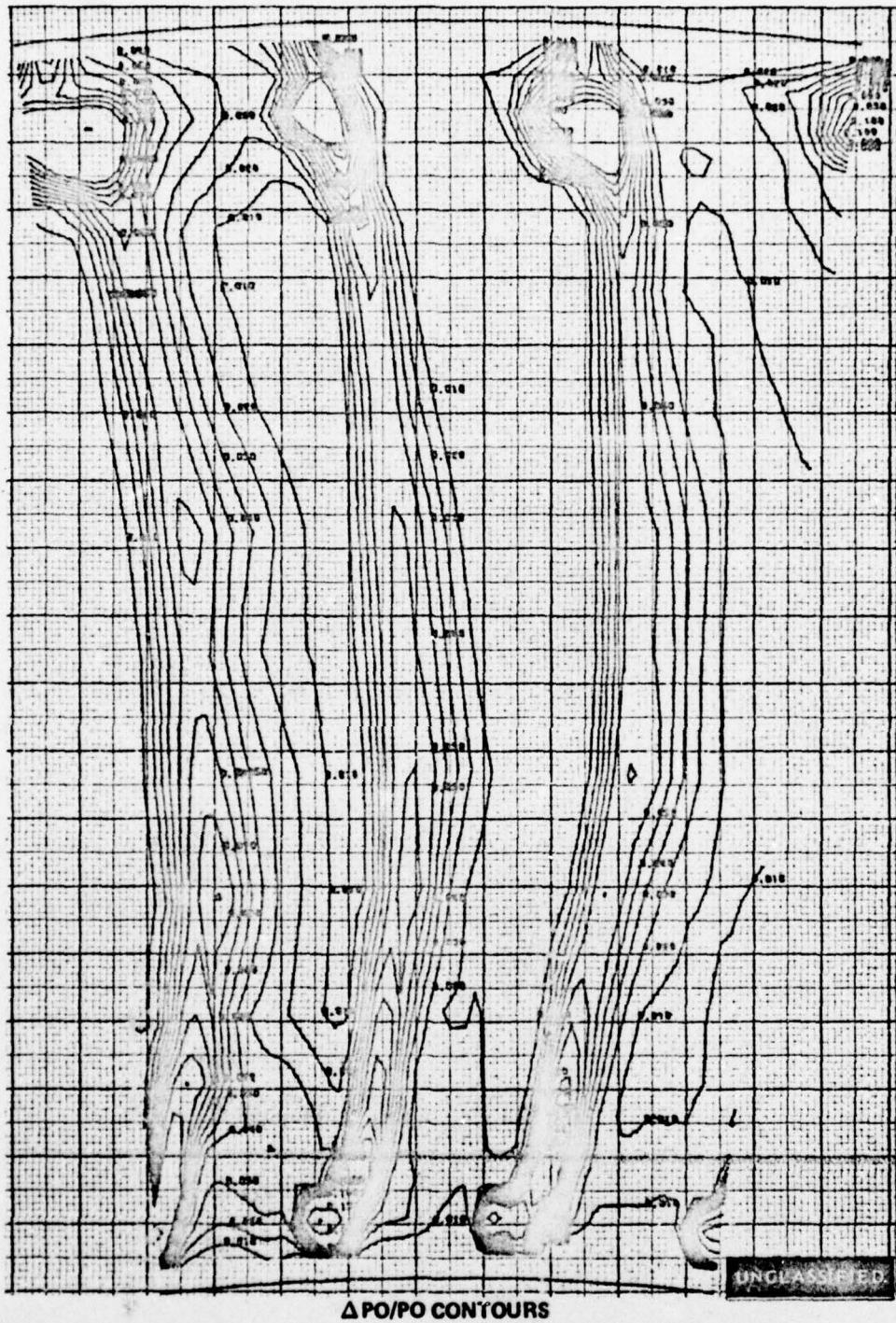


Figure 118 Spanwise Exit Mach Number Distribution, Second Vane Recambering Design  
B at Design Incidence. Midspan Exit Mach Number = 0.856

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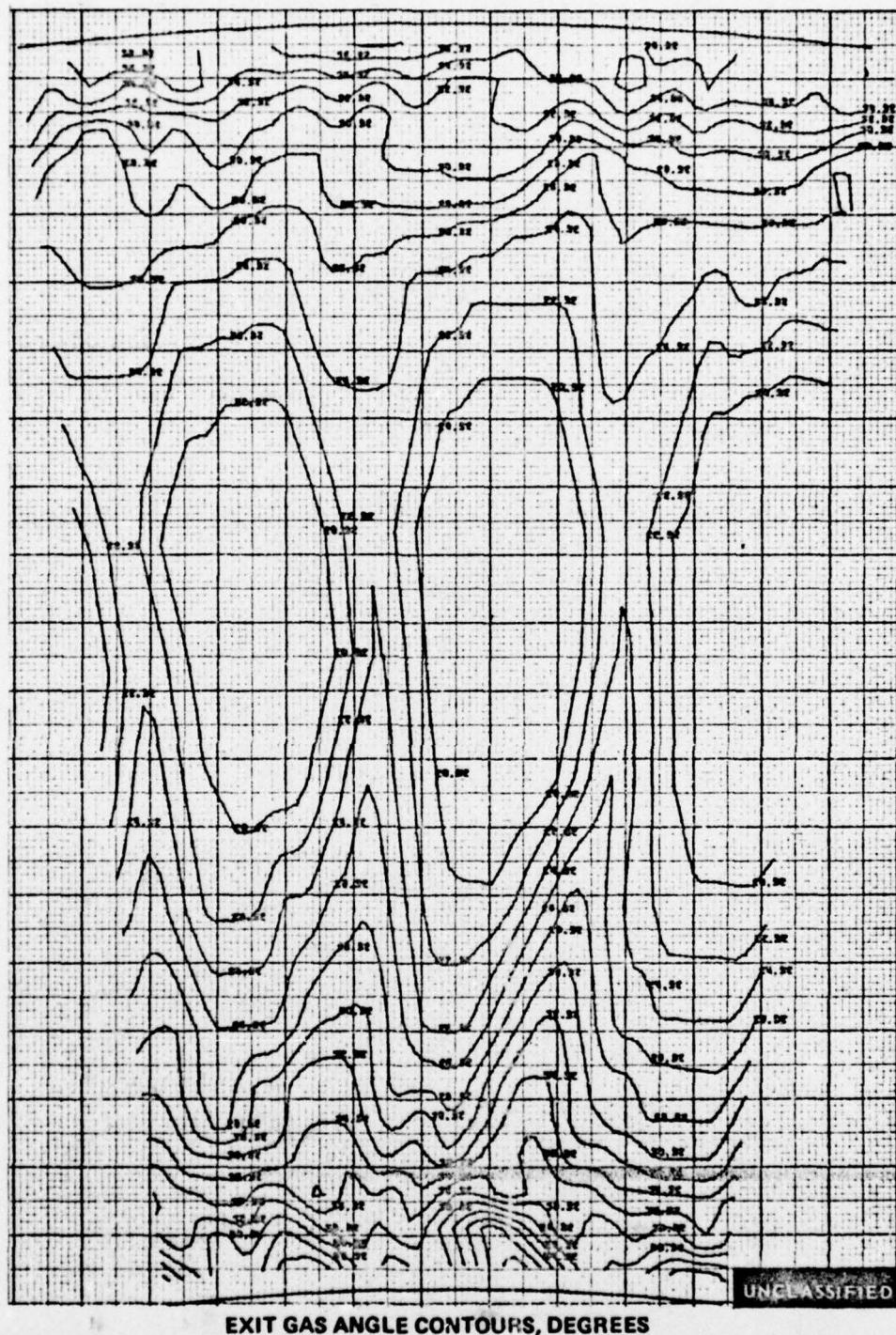
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**Figure 119 Pressure Loss Contours, Second Vane Recambering Design C at Design Incidence. Three Flow Passages. Midspan Exit Mach Number = 0.862**

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**Figure 120** Exit Gas Angle Contours, Second Vane Recambering Design C at Design Incidence. Three Flow Passages. Midspan Exit Mach Number = 0.862

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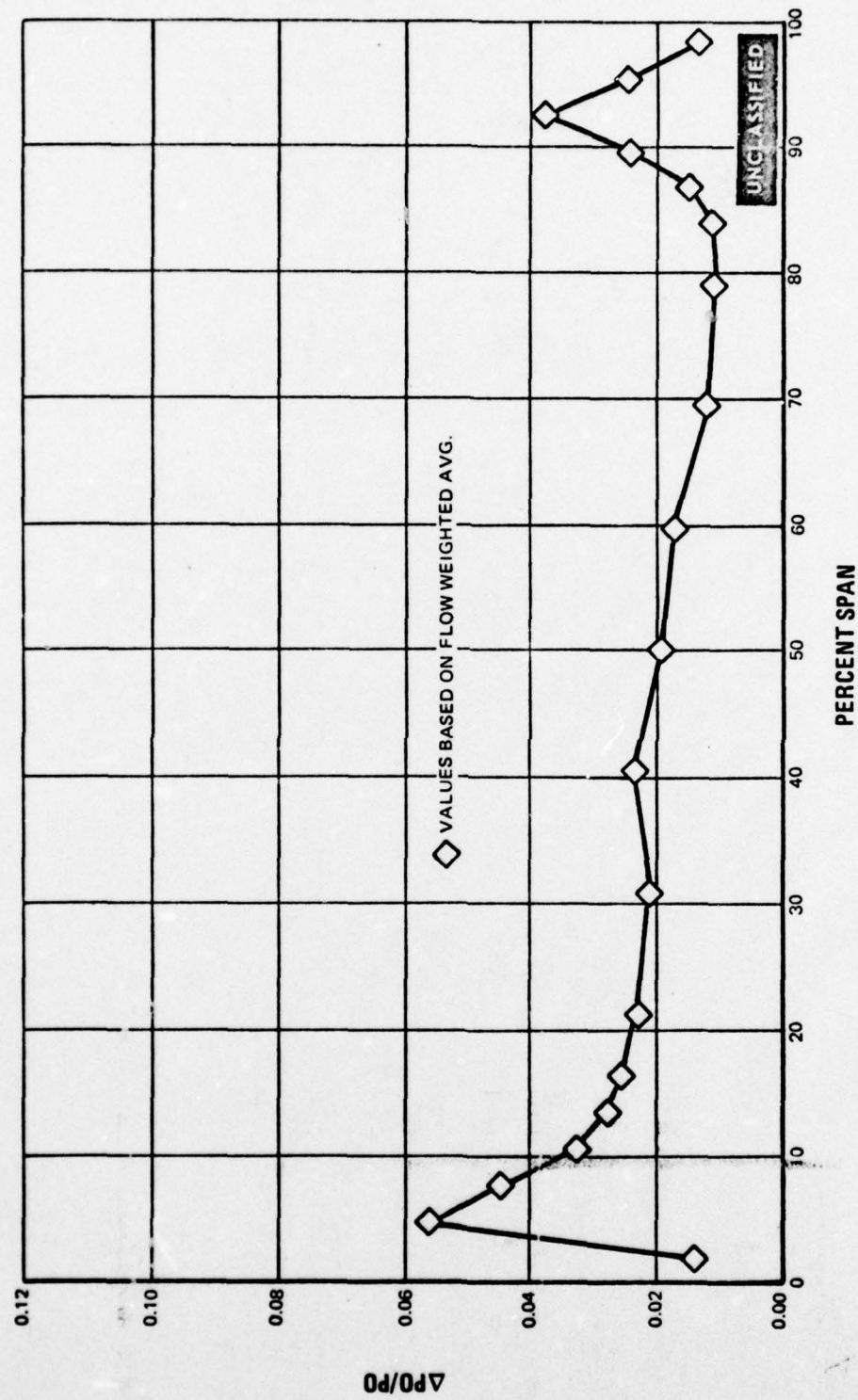


Figure 121 Spanwise Pressure Loss Distribution, Second Vane Recambering Design C  
at Design Incidence. Midspan Exit Mach Number = 0.862

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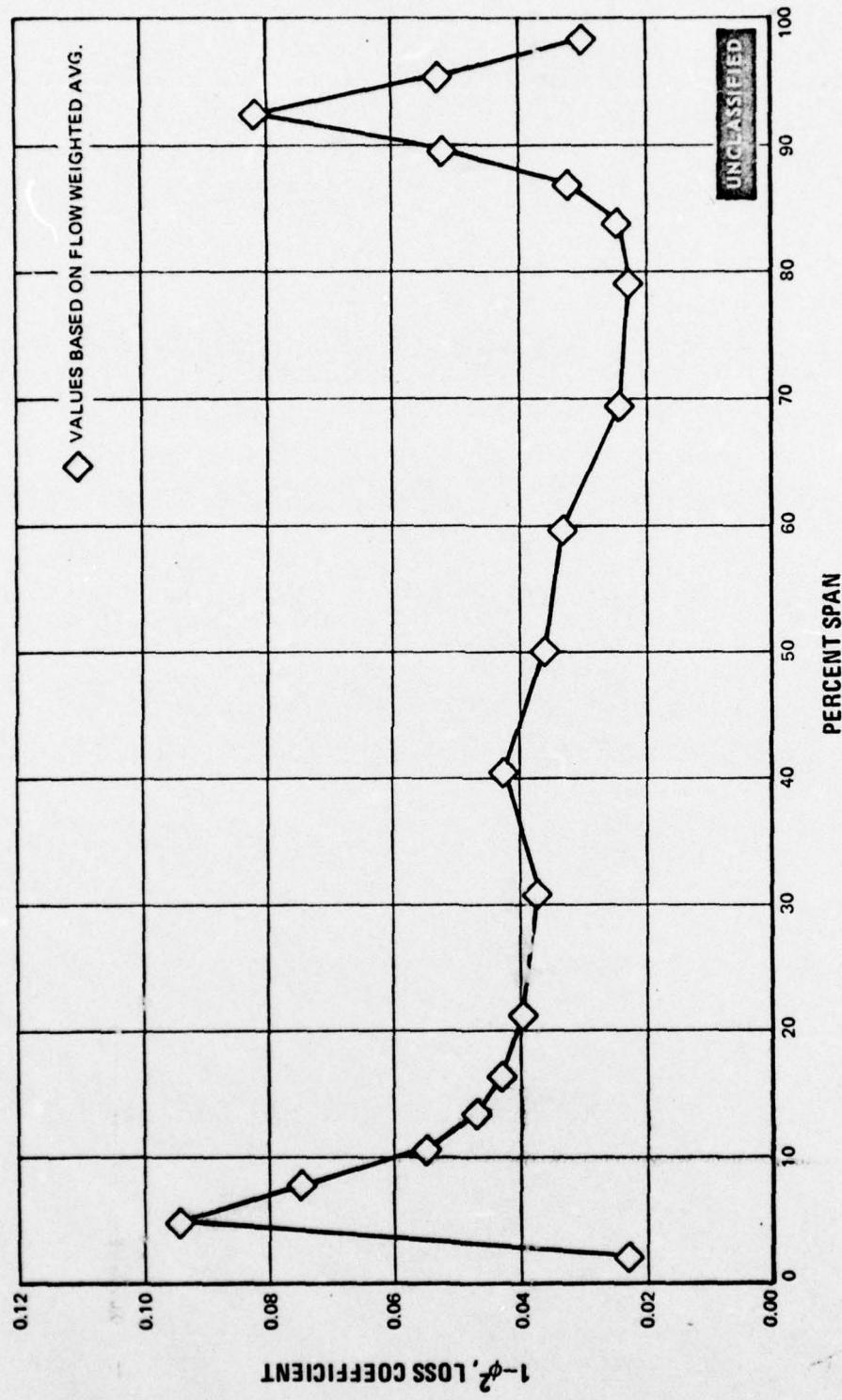


Figure 122 Spanwise Loss Coefficient Distribution, Second Vane Recambering Design C  
at Design Incidence. Midspan Exit Mach Number = 0.862

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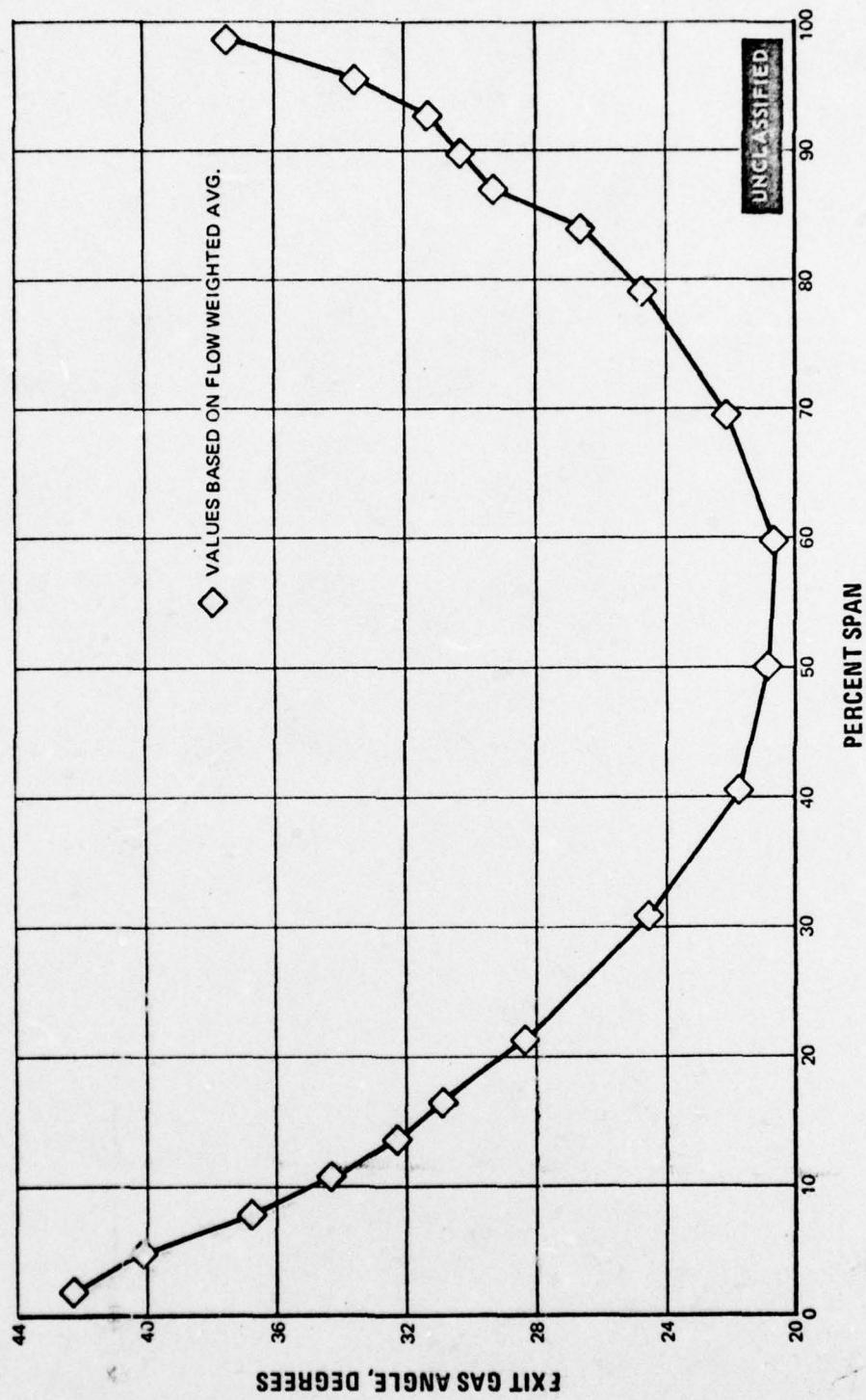


Figure 123 Spanwise Exit Gas Angle Distribution, Second Vane Recambering Design C  
at Design Incidence. Midspan Exit Mach Number = 0.862

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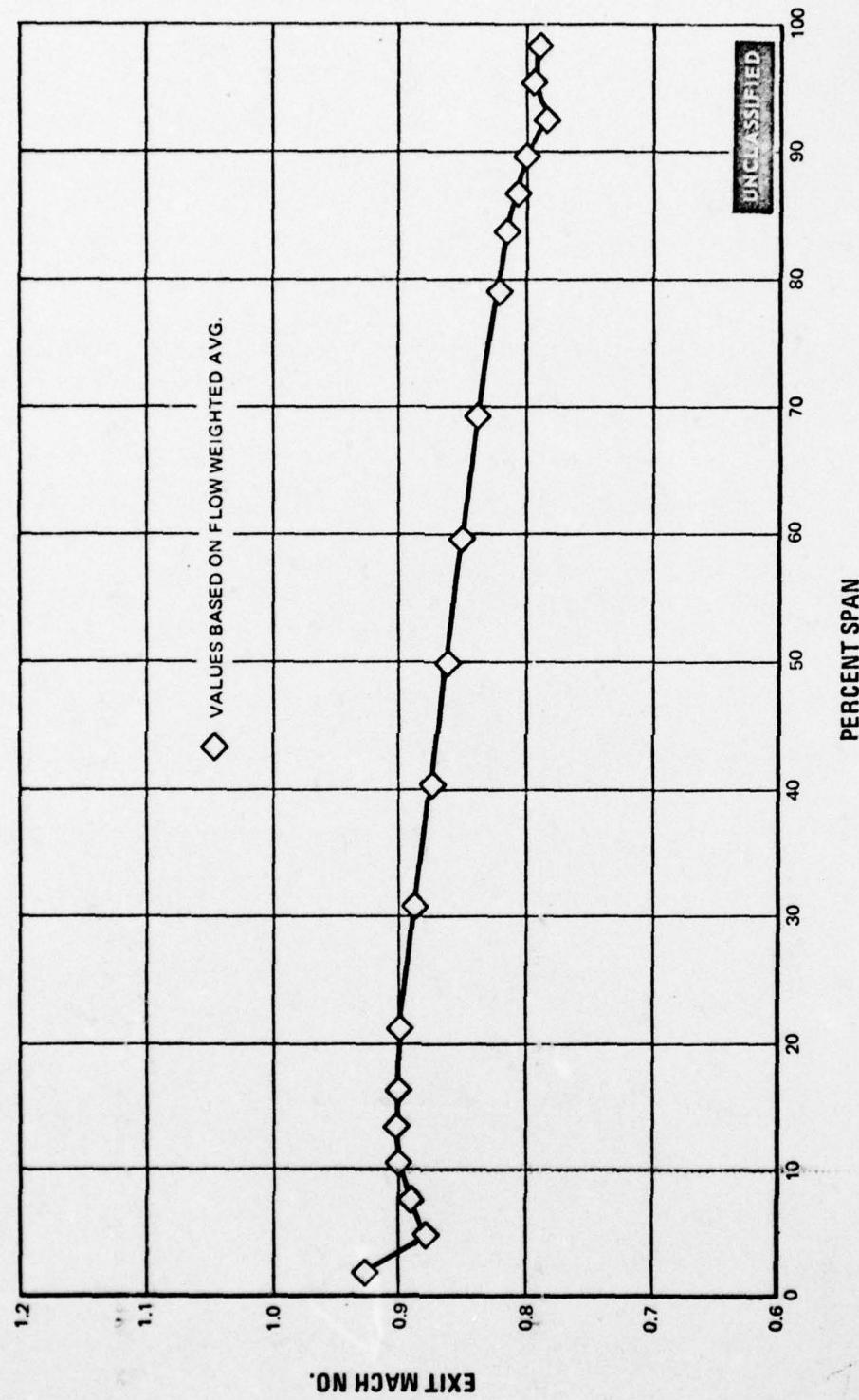


Figure 124 Spanwise Exit Mach Number Distribution, Second Vane Recambering Design C at Design Incidence. Midspan Exit Mach Number = 0.862

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(U) The spanwise loss coefficient distribution for recambering design A is shown in Figure 125. For comparison, the loss coefficient distributions are included for the second vane baseline airfoil and for the baseline airfoil after the first recambering was applied. The peak loss near the inside diameter of design A agreed well with that for the first recambered test, but the integrated midspan-to-root loss coefficient increased from 0.029 to 0.036. The midspan-to-tip integrated loss decreased from 0.038 for the first recambering to 0.035 for design A, reflecting the effect of opening the tip exit angle by approximately 10 degrees. The overall integrated loss coefficient indicated a slight increase (0.036) as compared to that for the first recambering (0.033). Losses at the midspan were close to the loss levels for the baseline airfoil

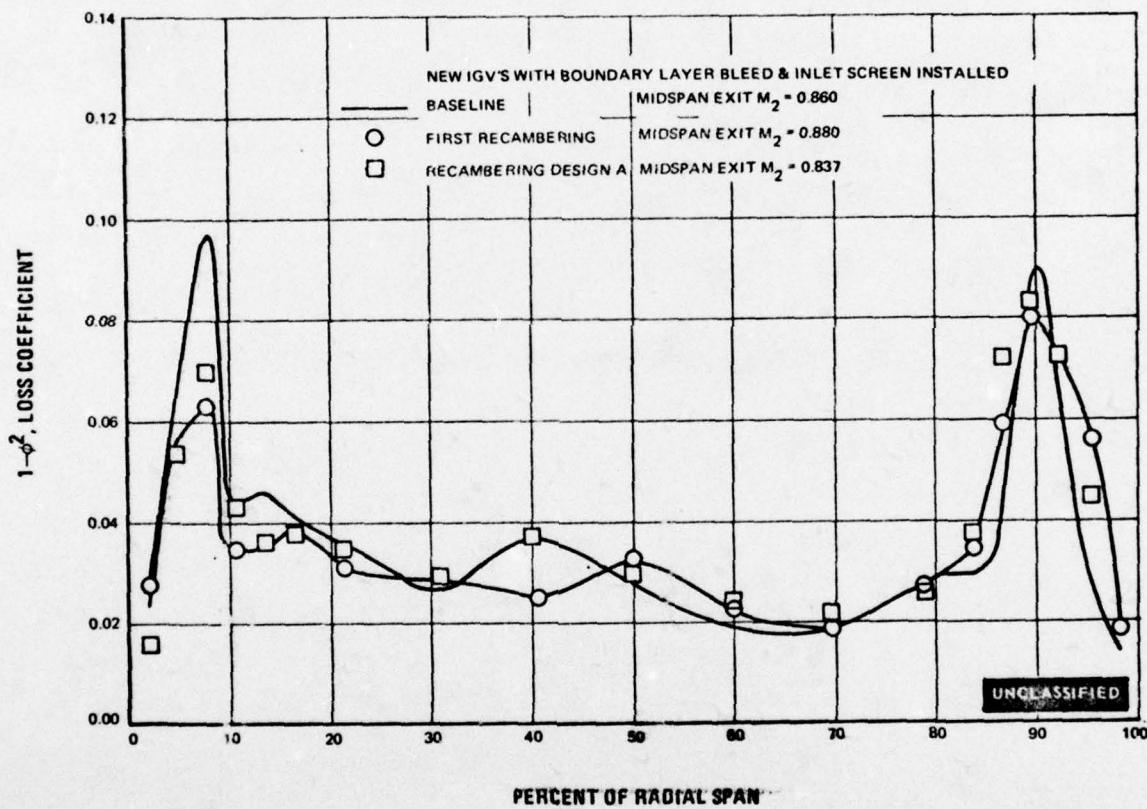


Figure 125 Spanwise Flow Weighted Loss Coefficient Distribution – Recambering Design A

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(U) As noted above, the differences between the measured loss values of these airfoils are very small. Therefore, the loss values that are quoted are integrated flow-weighted values, since it is difficult to evaluate the differences by observing the spanwise loss distributions. The quoted values are the midspan design exit Mach number, arrived at by extrapolation of plotted test data. These values will be compiled in Table XXXII in the Summary.

(U) The spanwise loss coefficient distribution for recambering design B is shown in Figure 126 for comparison with the baseline and first recambering. Although the root exit angle was increased by approximately 8.5 degrees from the baseline airfoil design, the peak loss at the root was identical to the baseline value. The integrated midspan-to-root loss coefficients were also essentially the same, since the loss coefficient for the baseline foil was 0.035, while for design B it was 0.031. This compares to 0.036 for design A and 0.029 for the first recambered airfoil. The design B exit tip angle was increased by about 10 degrees above that for the baseline value. Again, the peak and integrated losses for design B were almost identical to the baseline airfoil. The integrated midspan-to-tip loss coefficient for the baseline airfoil was 0.035, while for design B it was 0.033. This compares to 0.035 and 0.038 for design A and the first recambered airfoil, respectively.

(U) A similar comparison of loss coefficients is made for recambering design C in Figure 127. For this design, the root exit angle was increased approximately 13.5 degrees and the tip exit angle was increased by approximately 15 degrees over the baseline airfoil angles. The root losses for design C were almost identical to design B, and slightly greater than the baseline airfoil value. The tip losses were lower (0.034) than both the baseline (0.035) and design A (0.035) and higher than design B (0.033), with the peak shifting closer to the wall. Design C had the highest overall loss coefficient (0.038) when compared to the baseline (0.035), design A (0.0350) and design B (0.032). The midspan loss for design C was greater than the baseline airfoil design.

(U) The measured exit gas angles agreed well with the airfoil design angles for three recambering changes. These are presented in Figures 128, 129 and 130. Although the inlet flow angles were not completely defined by mapping the test channel (only a single radial traverse was made), the exit angle measurements indicate that the airfoils were turning the flow to the desired value.

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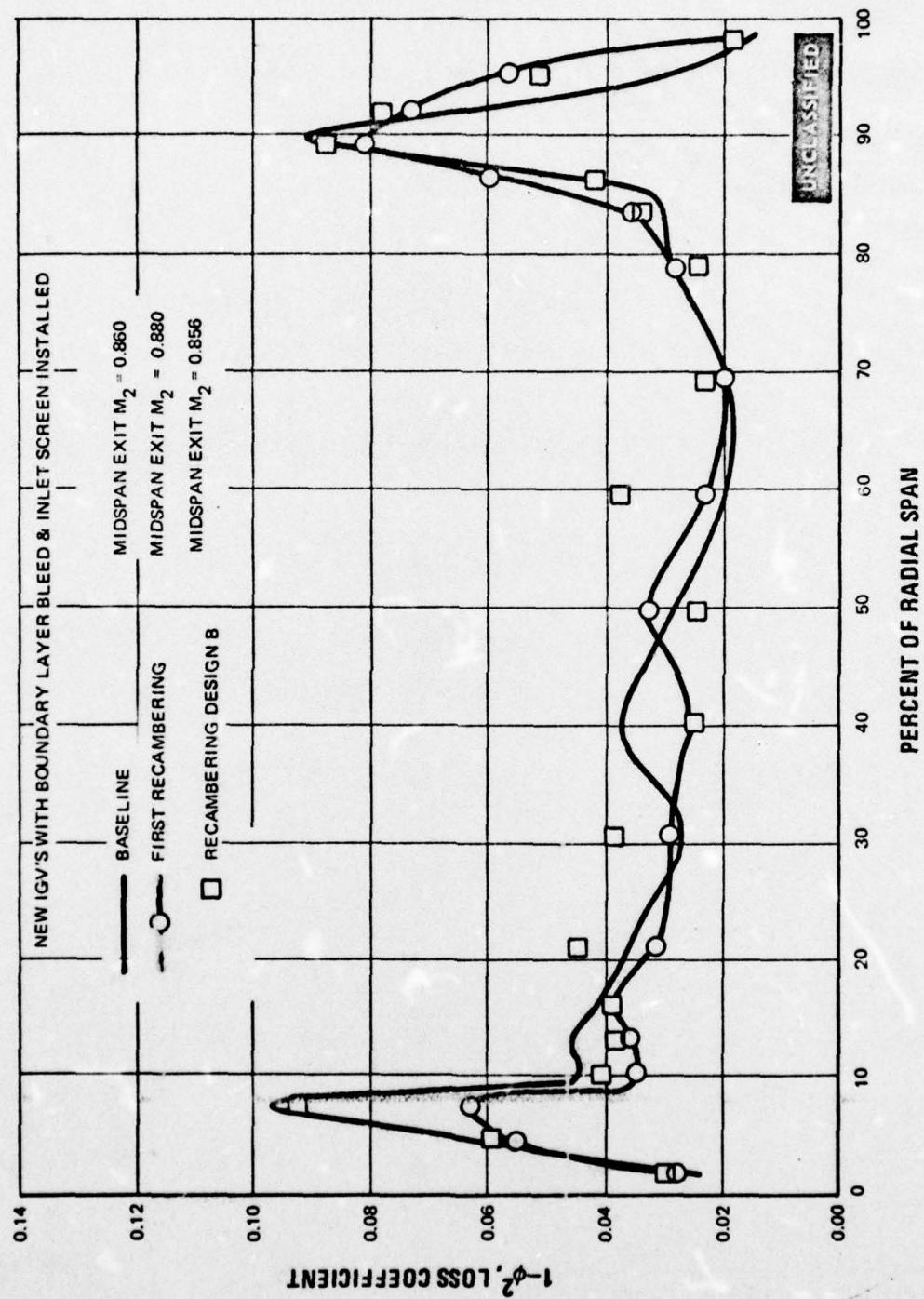


Figure 126 Spanwise Flow Weighted Loss Coefficient Distribution – Recambering Design B

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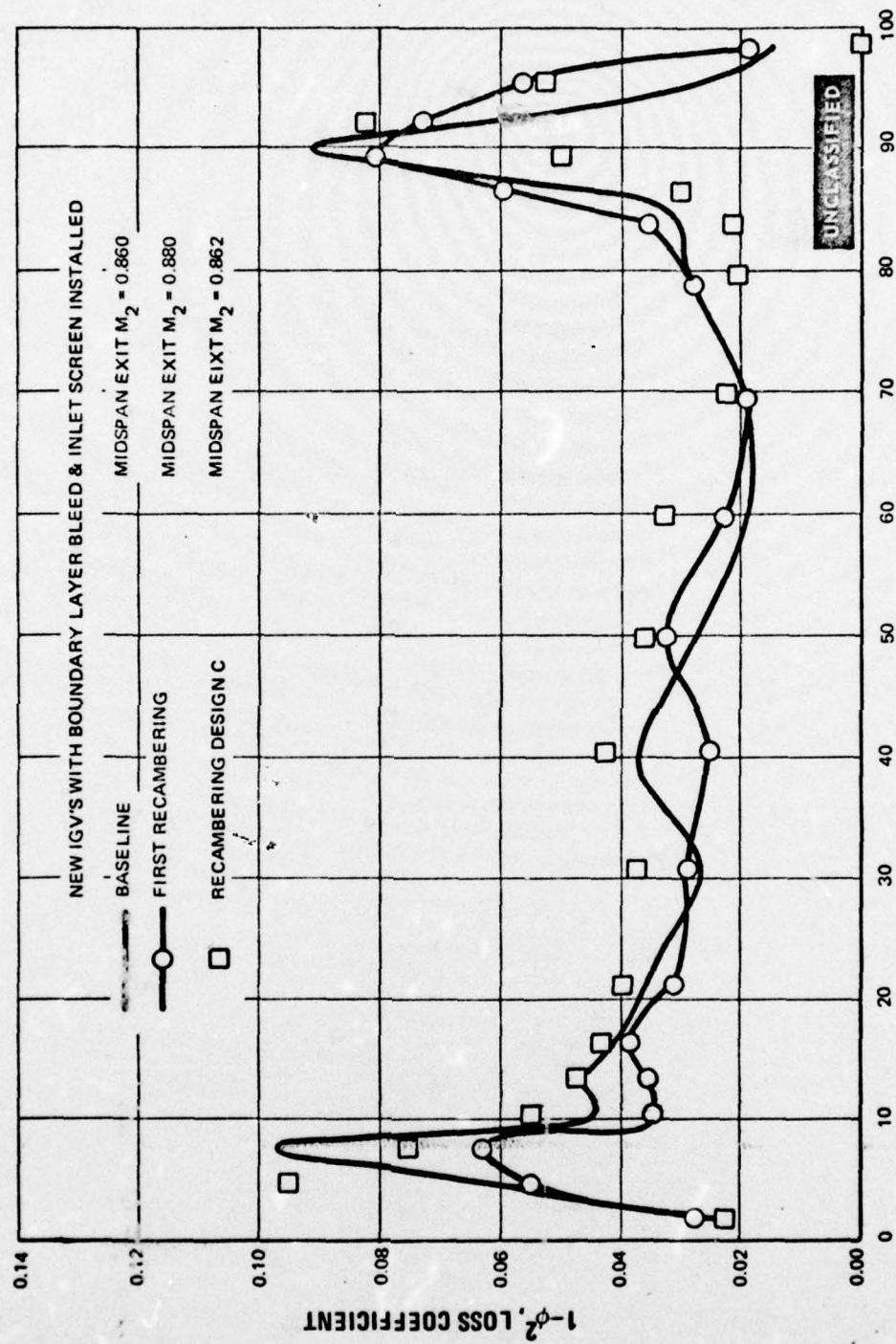


Figure 127 Spanwise Flow Weighted Loss Coefficient Distribution – Recambering Design C

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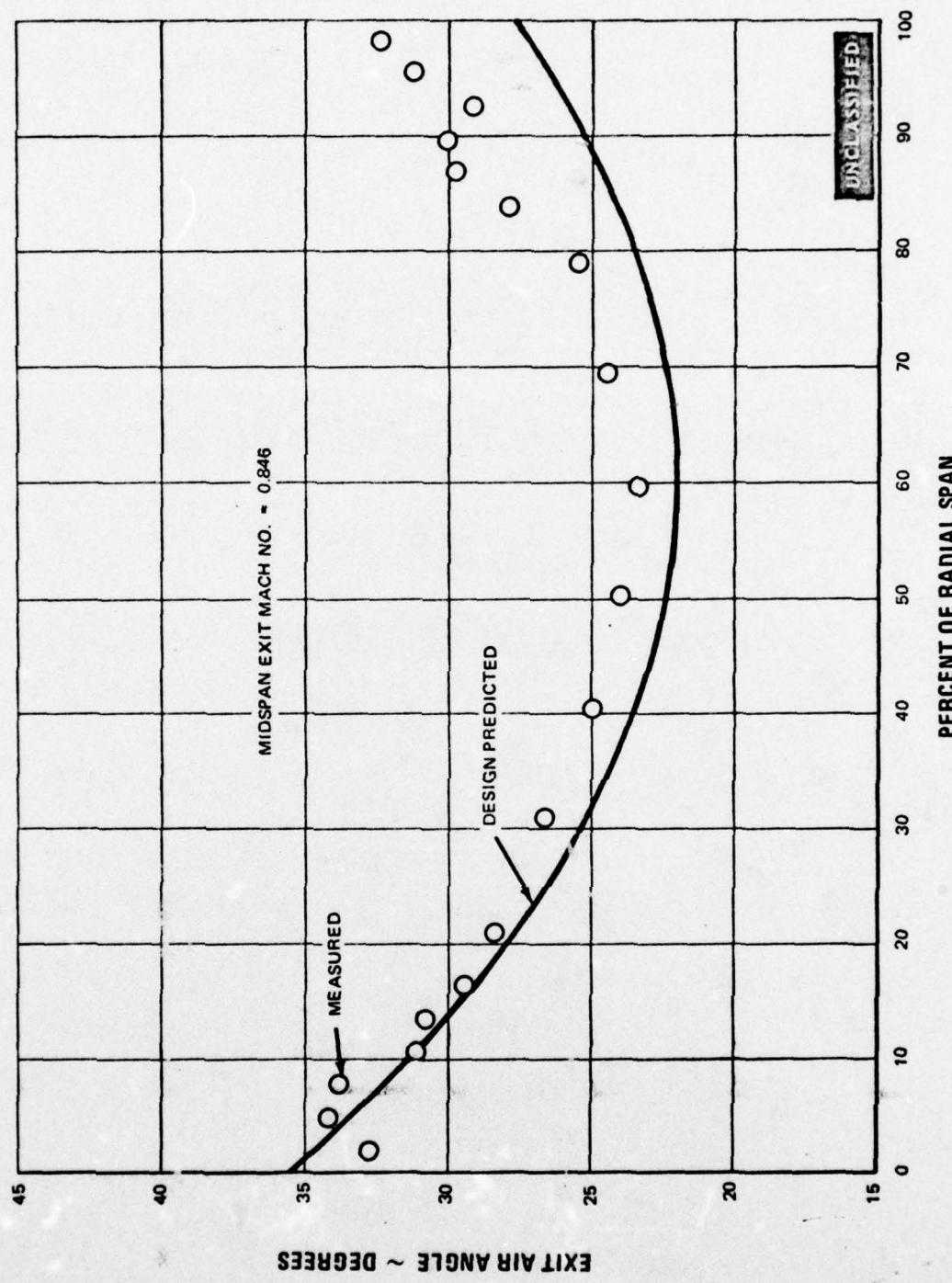


Figure 128 Measured Spanwise Exit Gas Angle Distribution – Recambeing Design A

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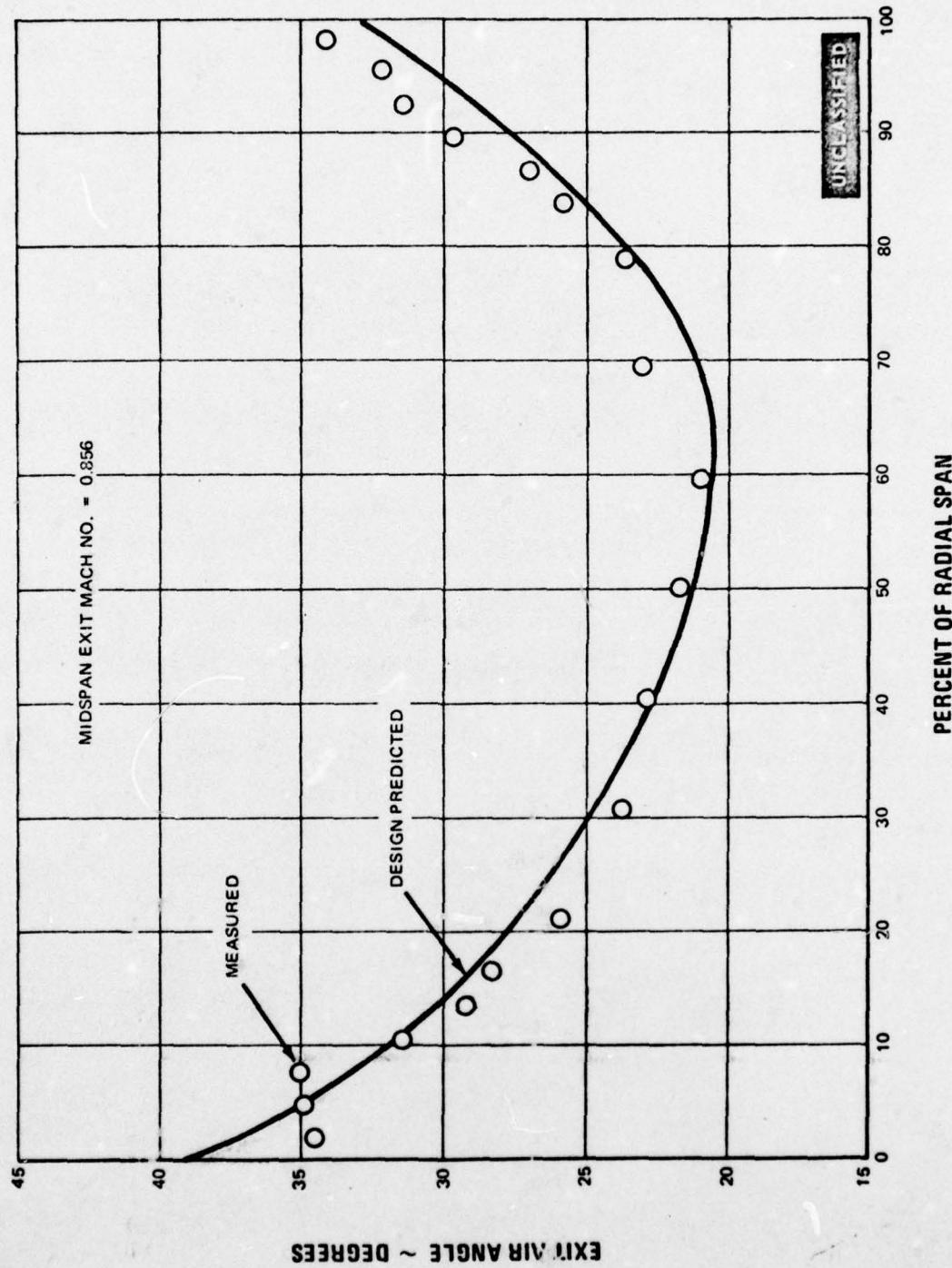


Figure 129 Measured Spanwise Exit Gas Angle Distribution – Recambering Design B

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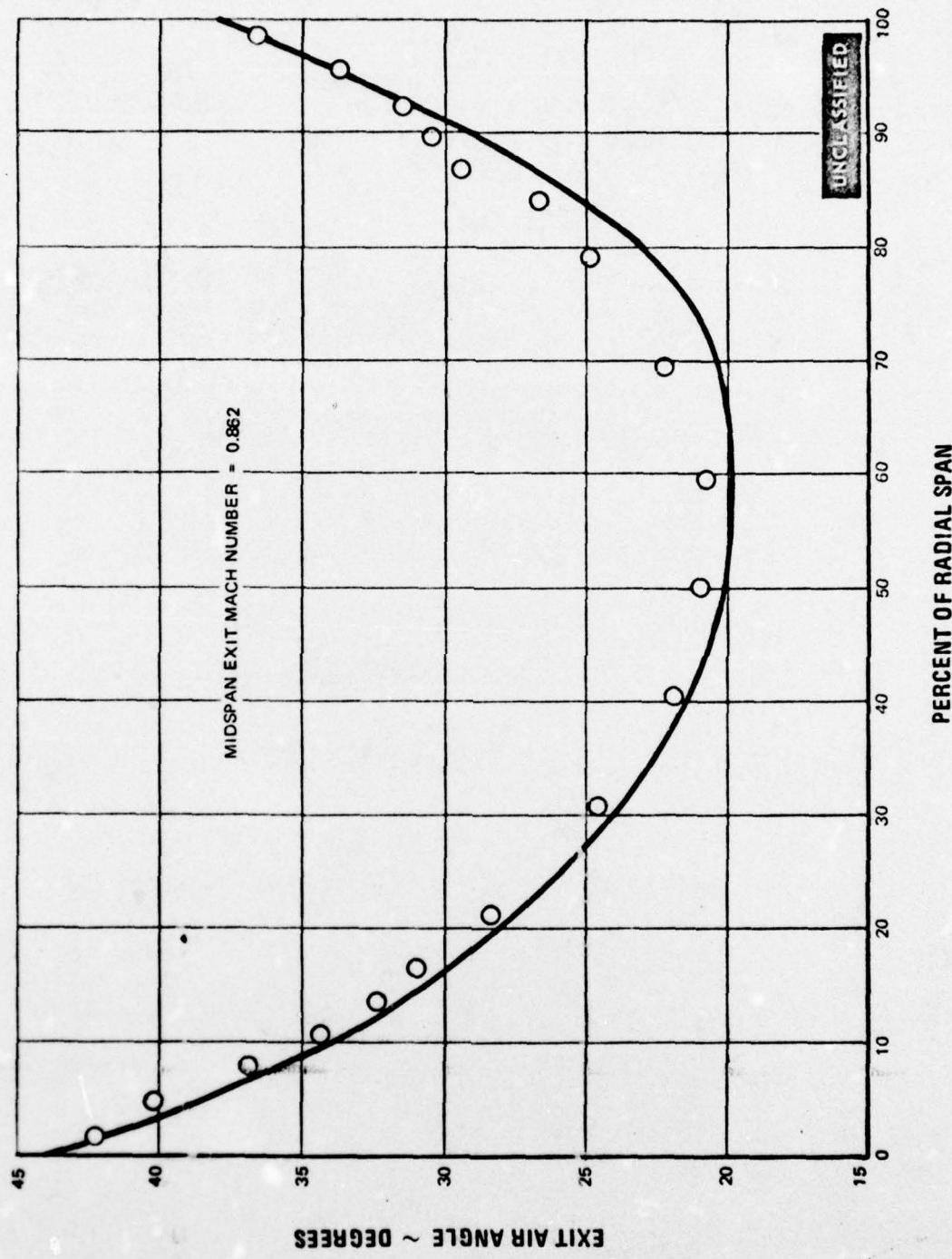


Figure 130 Measured Spanwise Exit Gas Angle Distribution – Recambering Design C

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Design A exhibited a few degrees of undeturning at the tip. The same order of magnitude of undeturning was observed for the baseline test in the same region (Reference 4). This undeturning disappears for designs B and C, indicating that better control of flow was obtained in the tip section.

(U) There was very small effect of Mach number variation on loss coefficient for the recambered airfoils. This is seen in Figure 131. None of the airfoils exhibited a transonic drag rise over the Mach number range tested. The Reynolds number was not held constant during these tests, and any variation of loss may be partly attributed to the fact that Reynolds number increased with Mach number in the annular segment cascade. The plane cascade tests of the baseline airfoil sections (Section III), where Reynolds number could be controlled independently of Mach number, indicated that there was only a small effect of Reynolds number on loss within the range of that investigation.

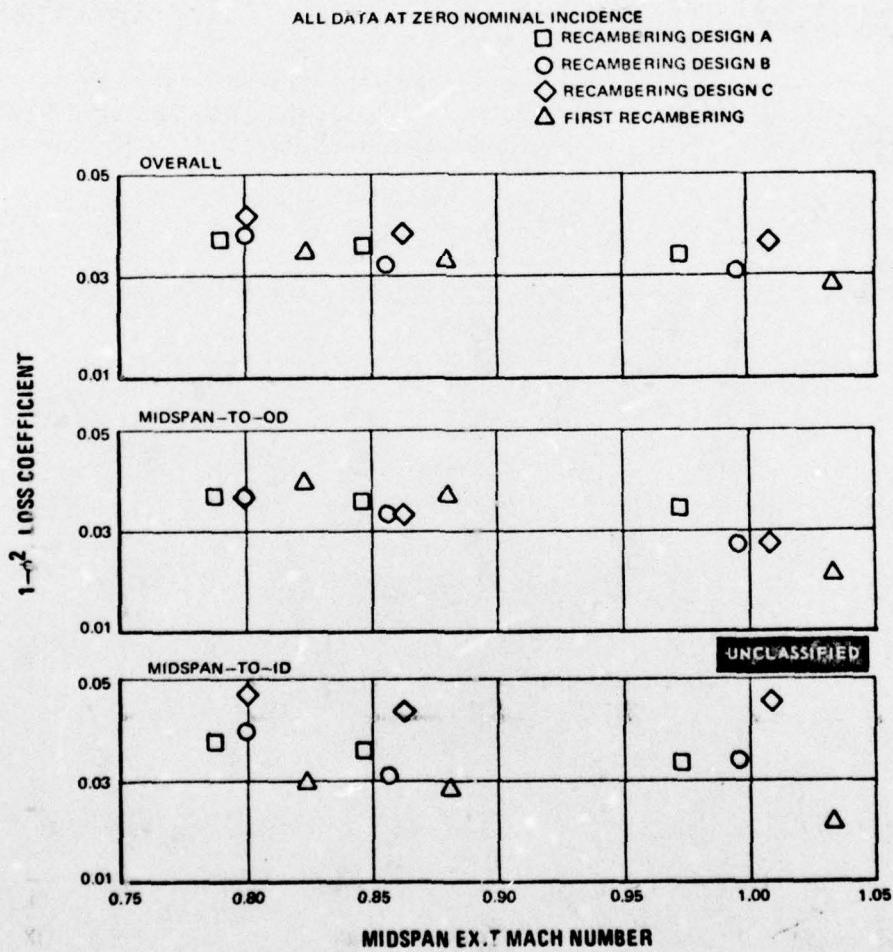


Figure 131 Effect of Mach Number on Loss for the Various Recambering Designs

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(U) As was done in the previous tasks, a mixture of oil and graphite was painted on the airfoils in order to provide additional information about the behavior of the flow in these recambered airfoil cascades. Photographs of the resulting patterns were taken and are shown in Figures 132 and 133 for recambering design A, in Figures 134 and 135 for design B, and in Figures 136 and 137 for design C. These patterns indicate that the flow remained attached on the airfoils and end-walls. Cross-flow patterns can be seen from these photographs at the walls, but no separation on these walls in the vicinity of the airfoils was detected.

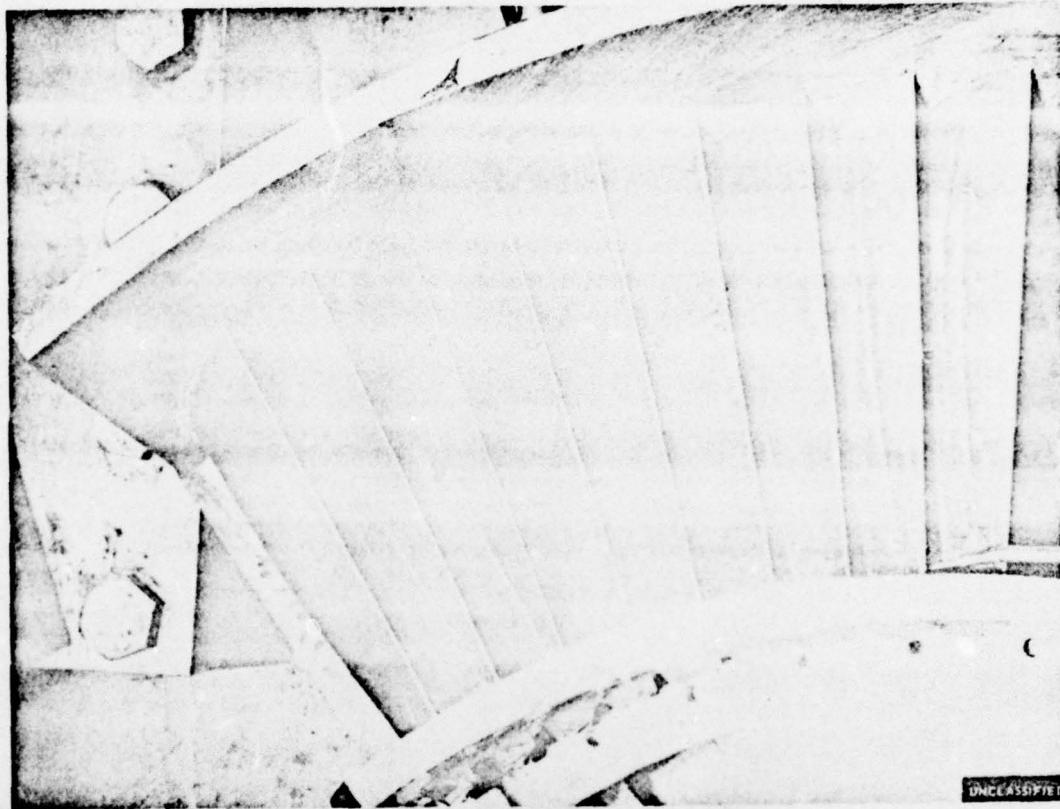


Figure 132 Oil and Graphite Flow Patterns, Second Vane Recambering Design A, Mach No. = 0.846 (FE95157)

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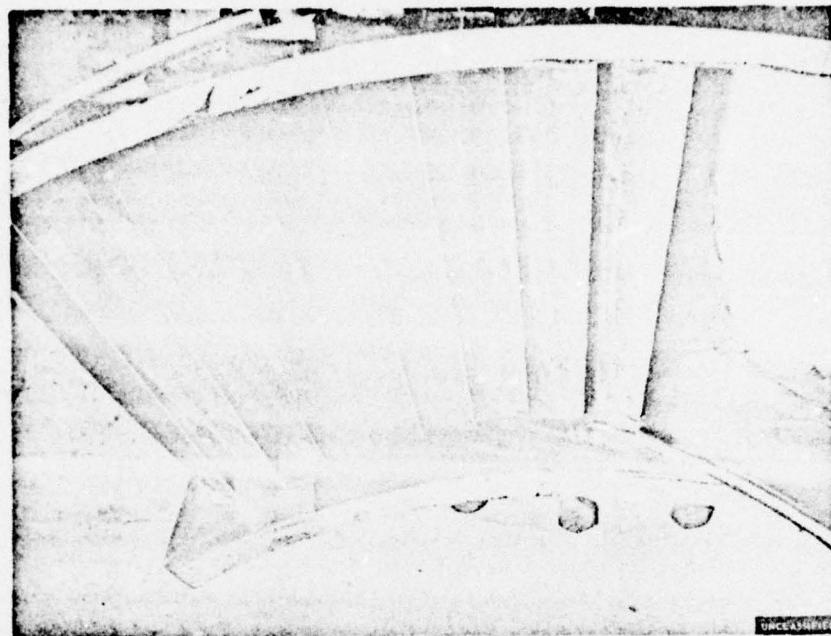


Figure 133 Oil and Graphite Flow Patterns, Second Vane Recambering Design A, Mach No. = 0.846 (FE95158)

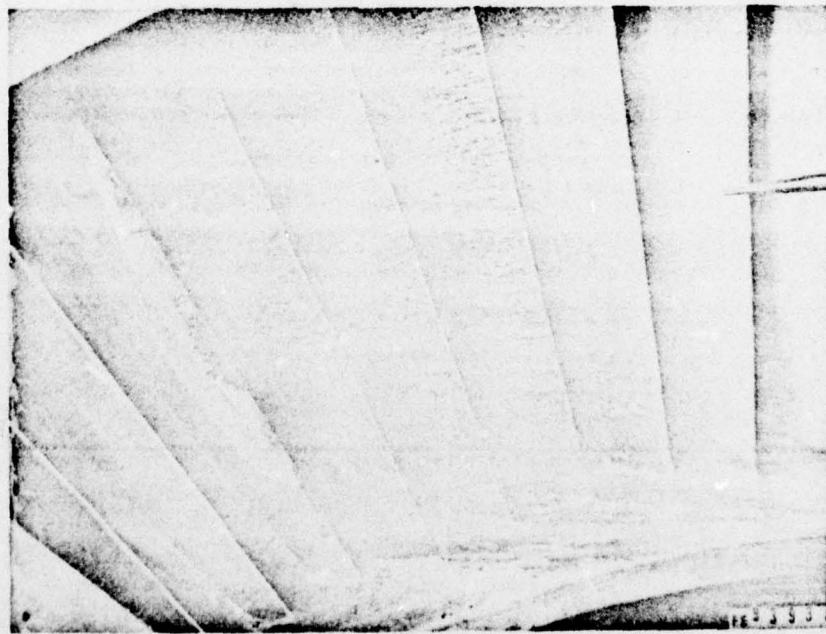


Figure 134 Oil and Graphite Flow Patterns, Second Vane Recambering Design B, Mach No. = 0.856 (FE93931)

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Figure 135 Oil and Graphite Flow Patterns, Second Vane Recambering Design B, Mach No. = 0.856 (FE 93939)

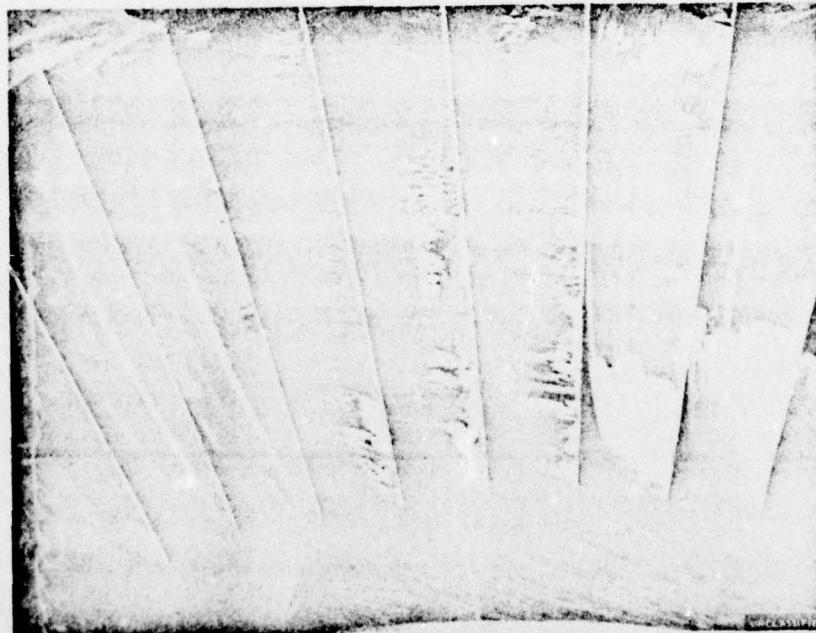


Figure 136 Oil and Graphite Flow Patterns, Second Vane Recambering Design C, Mach No. = 0.862 (FE95165)

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Figure 137 Oil and Graphite Flow Patterns, Second Vane Recambering Design C, Mach No. = 0.862 (FE95166)

## 5. SUMMARY

(U) Results of the second vane recambering test evaluation at zero incidence and at nominal design exit Mach number are tabulated in Table XXXII. During these tests, three levels (approximately 5, 10 and 15 degrees) of local uncambering at the end-walls were investigated. With such large changes in local airfoil geometry, the end-wall losses could not be substantially reduced. The 10 degree uncambered cascade (Design B) had a slightly lower overall loss than the 5 and 15 degree uncambered cascades (Designs A and C, respectively). The root end loss was the lowest for recambering design B with small differences in tip end loss for the three recambered designs. Midspan profile losses increased as expected with an increasing amount of midspan overcambering applied in each of the three cascades to maintain constant overall work. The exit flow angles for recambering designs A, B and C behaved in a predictable manner, indicating that such uncambering can be a potentially useful technique for improving entrance flow conditions for the following row of airfoils.

(U) This Task IIIa information will be applied in the demonstrator turbine design (Phase IV). Data shows that the overall loss coefficient for the same overall work was only slightly affected by a large redistribution of turning. Furthermore, the data indicates that a slight reduction of end-wall loss results from a small amount of the type of recambering that was investigated. As a result, i.e final design studies (Phase IV) will attempt to gain some advantage from the direct and indirect benefits of such recambering. The indirect benefits include better flow control and better spanwise distribution.

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TABLE XXXII

COMPARISON OF LOSS COEFFICIENTS FOR SECOND VANE RECAMBERING  
CORRECTED TO DESIGN POINT MACH NUMBER = 0.869  
ZERO INCIDENCE

Airfoil	MEASURED LOSS COEFFICIENT, $1-\phi^2$		
	Midspan-To-Root	Midspan-To-Tip	Overall
Baseline	0.035	0.035	0.035
First Recambering	0.029	0.038	0.033
Recambering Design A	0.036	0.035	0.036
Recambering Design B	0.031	0.033	0.032
Recambering Design C	0.044	0.034	0.038

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## 6. TEST PROCEDURE

(U) The test procedure employed during this task was identical to that of the Task II b baseline evaluation. This procedure was described in complete detail in Reference 3.

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## SECTION VI

### OFF-DESIGN EVALUATION (TASK IIIb AND IIIc)

#### 1. RFP OBJECTIVE

(U) Determine the effect of off-design operation on the aerodynamic performance of the selected airfoil geometries having the optimum boundary control method applied.

#### 2. TASK OBJECTIVE

(U) The purpose of these tasks was to evaluate off-design performance of the airfoil geometries which showed optimum end-wall losses during the Task IIIa investigation. These cascades were tested at three Mach numbers, including the design point, and at positive and negative incidence angles. The incidence angle was varied by changing the inlet guide vane geometry. Exit flow survey data were obtained and local and integrated losses and flow angle deviation were evaluated. This data will provide the basis for the airfoil loss system which will be integrated with the turbine design system.

#### 3. CASCADE PACK DESIGN

(U) New turning vane assemblies were fabricated for the off-design evaluations. Initially, it was decided to test at  $\pm 10$  degree incidence. However, in order to build the  $\pm 10$  degree incidence inlet guide vane, a new inlet bellmouth would have to have been fabricated. The largest negative incidence guide vane which could have been fabricated within the second vane bellmouth limitations was -6 degrees. This, therefore, was the negative incidence at which the tests were run. There was no problem in attaining the +10 degrees incidence inlet guide vanes for the bellmouth design on hand.

#### 4. DISCUSSION

(U) Four airfoils were tested at incidence angles of +10 degrees and -6 degrees. These airfoils included the second vane baseline airfoil, the second vane with first recambering, re cambering design A, and recambering design C. The results of these off-design tests will be discussed in the same order, and later tabulated in the Summary (Table XXXIII).

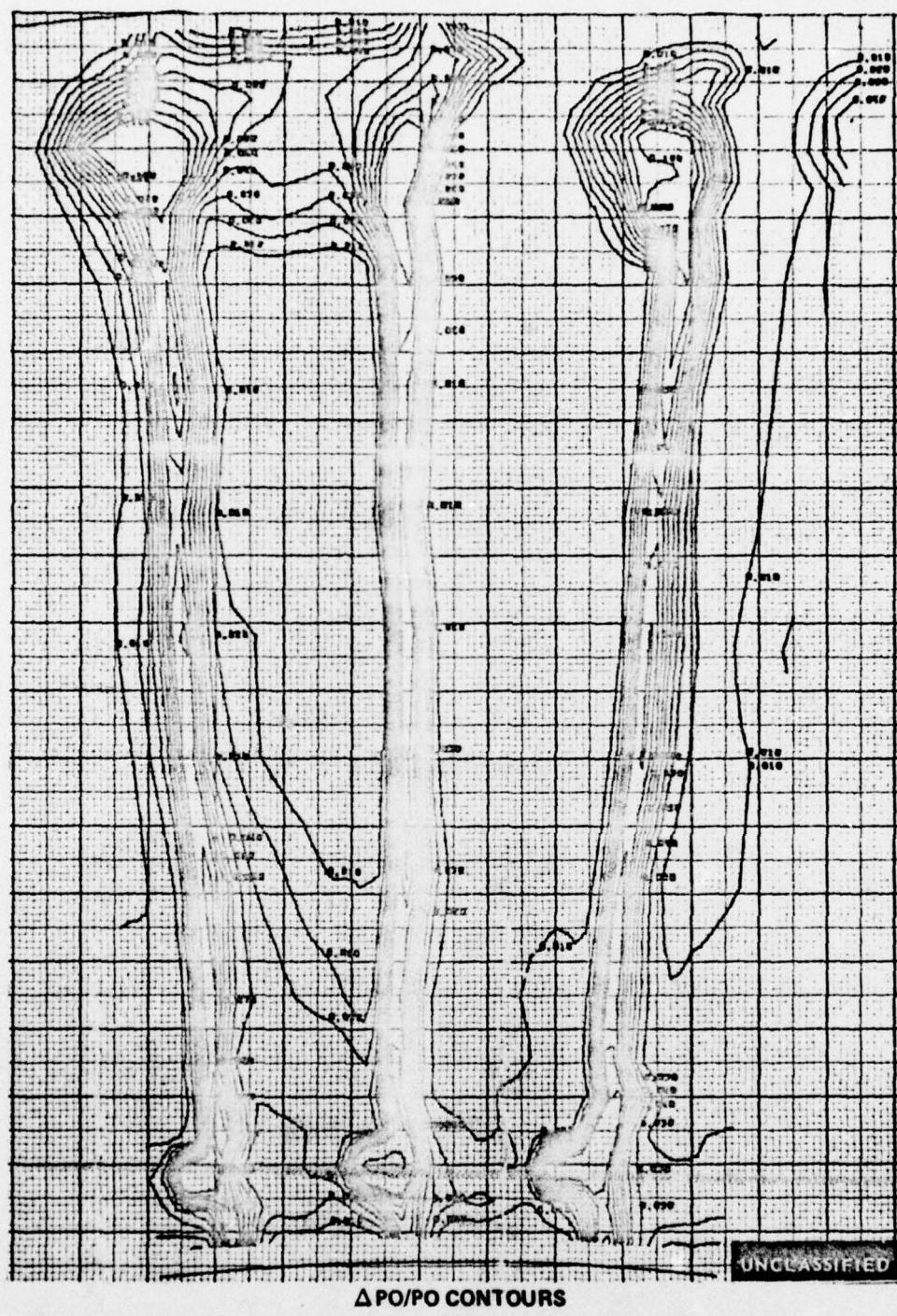
(U) The baseline airfoil performance data at an incidence angle of +10 degrees are presented for an exit midspan Mach number of 0.835 in Figures 138 through 143. Care was taken to choose a passage that was free of inlet guide vane wakes for spanwise evaluations. As in Reference 4, the following is included whenever "performance data" is referred to in this Report:

- total pressure loss contour plots
- exit gas angle contour plots
- average spanwise total pressure loss distribution
- average spanwise loss coefficient distribution
- average spanwise exit flow angle distribution
- average spanwise exit Mach number distribution

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- (U) The performance data are presented for an incidence angle of -6 degrees at a mid-span exit Mach number of 0.865 in Figures 144 through 149. Baseline airfoil comparisons of the spanwise loss coefficient distribution at +10 degrees incidence with zero incidence and -6 degrees incidence with zero incidence are shown in Figures 150 and 151, respectively. The root end-wall losses were greater at the positive incidence, and less at the negative incidence than the zero incidence values. The tip end-wall losses at the positive incidence were almost identical at the zero incidence, and again less than zero incidence for the negative incidence condition. On an overall integrated basis, the positive incidence loss coefficient was greater, and the negative incidence loss coefficient was less than the zero incidence value. The measured exit gas angles for positive and negative incidence are shown in Figures 152 and 153, respectively. These results indicate that the measured angles were, in general, in close agreement with the predicted values as was the case for previous tests (Task III a). Oil and graphite flow visualization tests were made on the test airfoils at both incidence angles and these are shown in Figures 154 through 157. These photographs indicate attached flow over the airfoil and end-walls and, as in the zero incidence case, some radial inflow at the end-walls due to the usual suction surface corner vortex.
- (U) Performance data for the second vane first recambered airfoil are shown in Figures 158 through 163 at the positive incidence angle of +10 degrees. Similar data for the negative incidence of -6 degrees are shown in Figures 164 through 169.

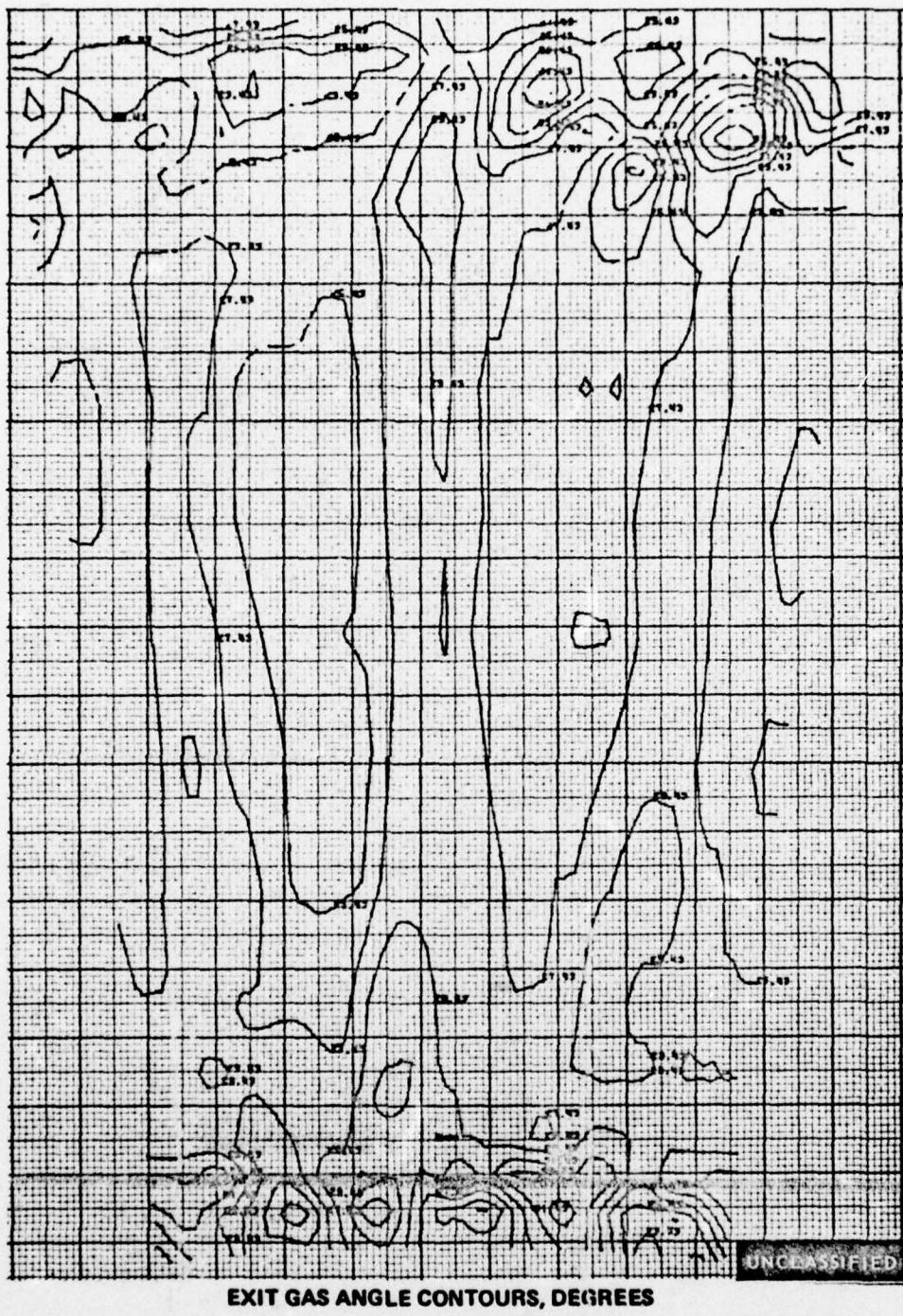
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**Figure 138 Pressure Loss Contours, Second Vane Baseline at  $+10^\circ$  Incidence. Three Flow Passages. Midspan Exit Mach Number - 0.835**

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**Figure 139** Exit Gas Angle Contours, Second Vane Baseline at  $+10^\circ$  Incidence. Three Flow Passages. Midspan Exit Mach Number = 0.835

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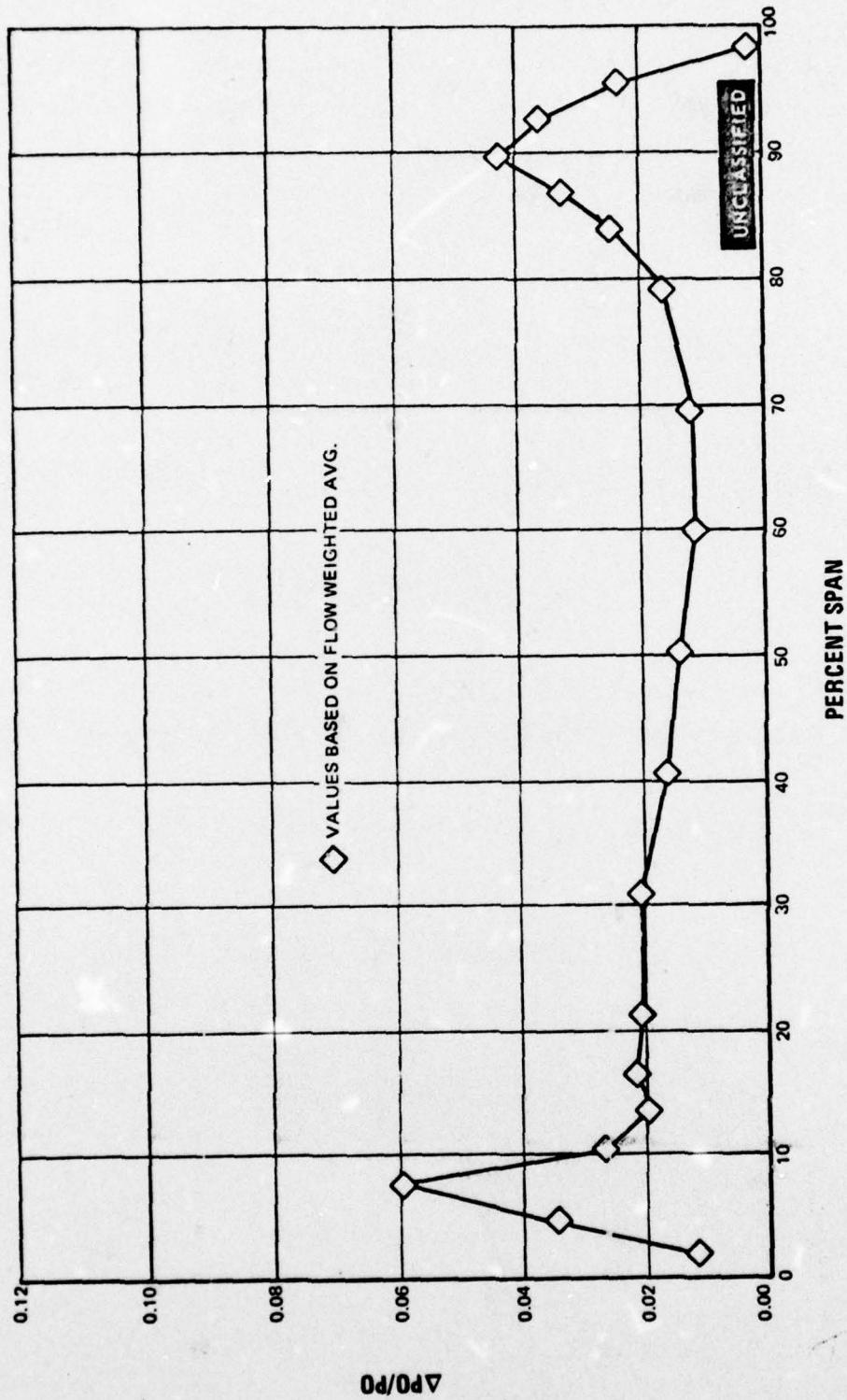


Figure 140 Spanwise Pressure Loss Distribution, Second Vane Baseline at +10° Incidence.  
Midspan Exit Mach Number = 0.835

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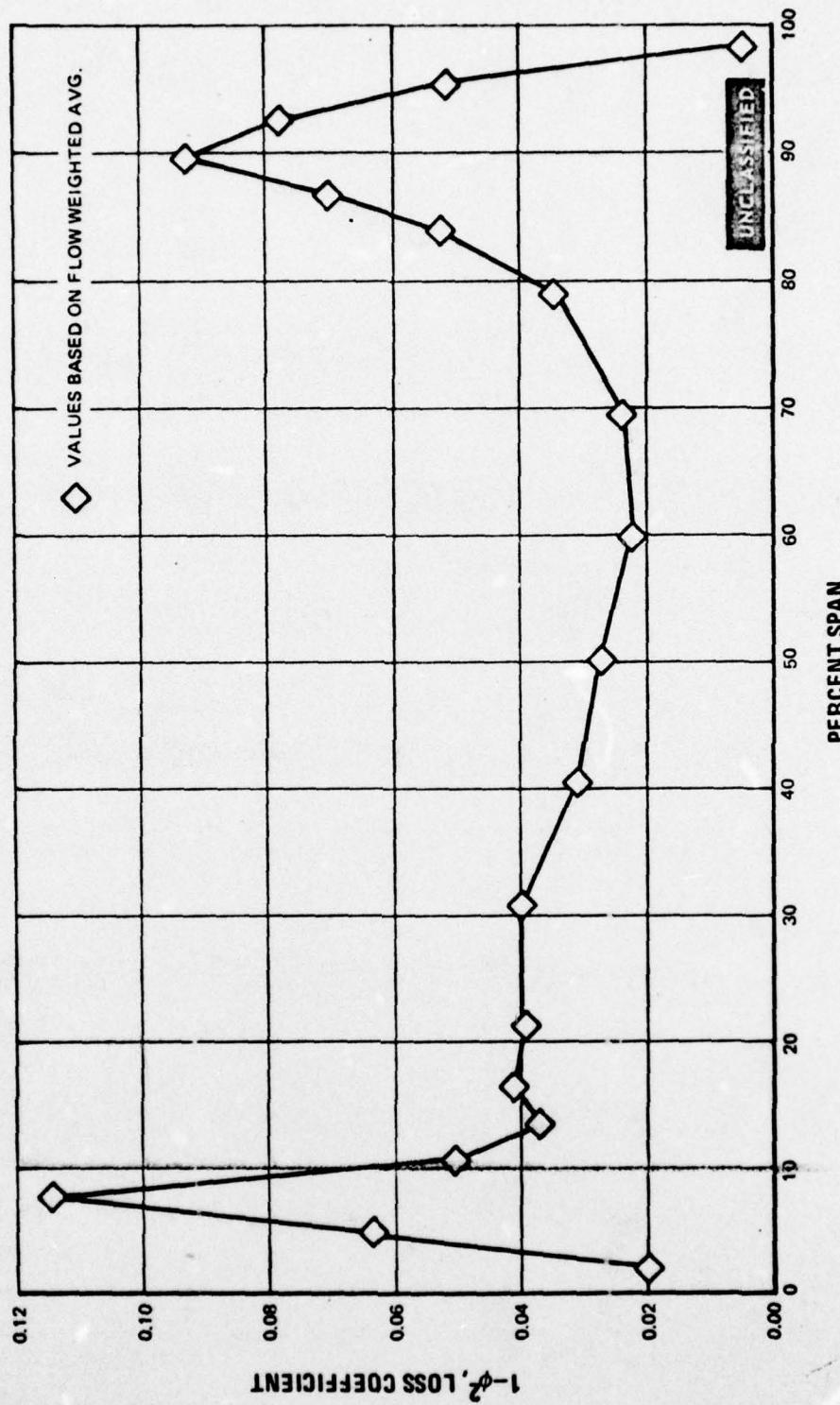


Figure 141 Spanwise Loss Coefficient Distribution, Second Vane Baseline at +10° Incidence. Midspan Exit Mach Number = 0.835

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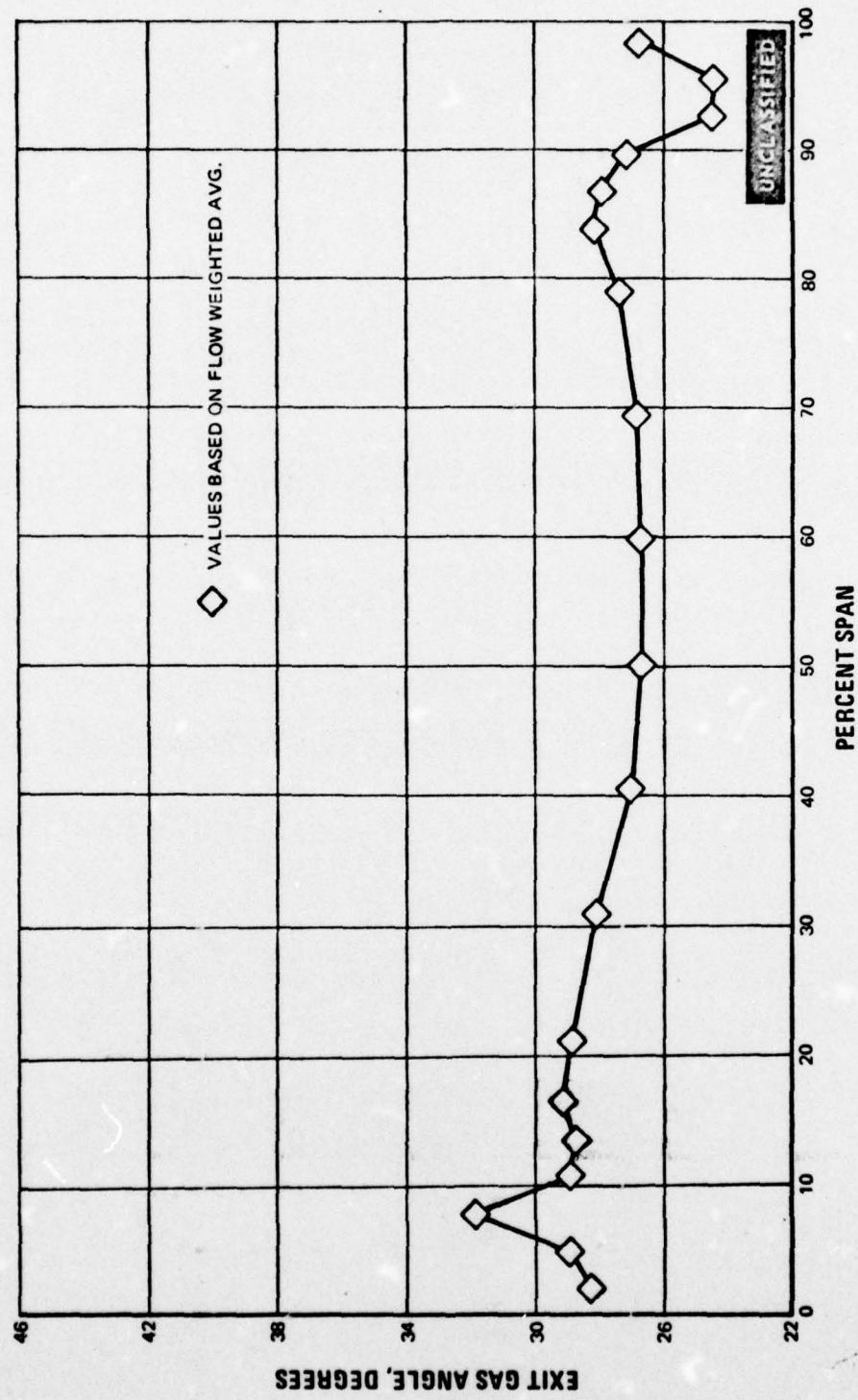


Figure 142 Spanwise Exit Gas Angle Distribution, Second Vane Baseline at  $+10^\circ$  Incidence. Midspan Exit Mach Number = 0.835

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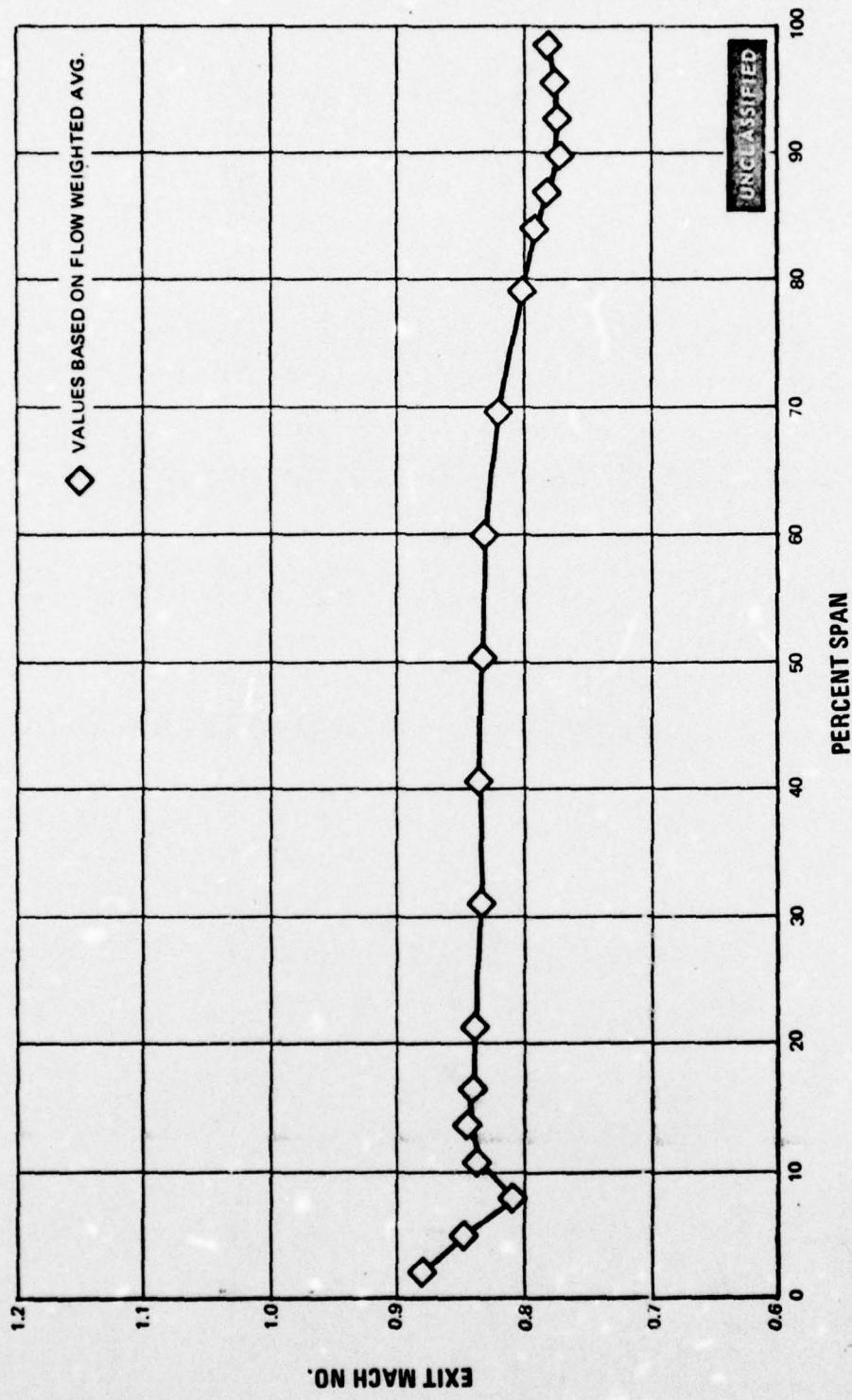
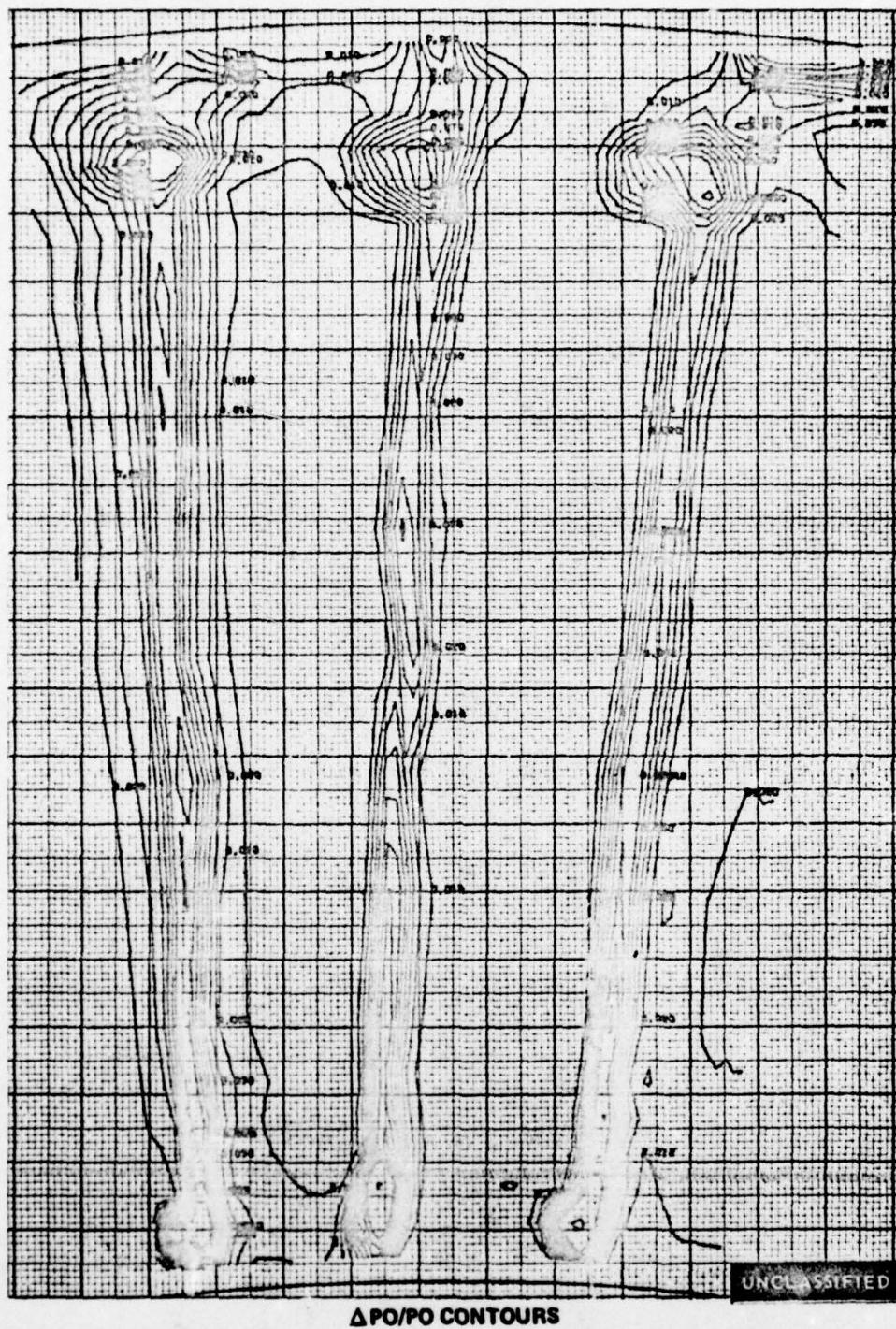


Figure 143 Spanwise Exit Mach Number Distribution, Second Vane Baseline at +10° Incidence. Midspan Exit Mach Number = 0.835

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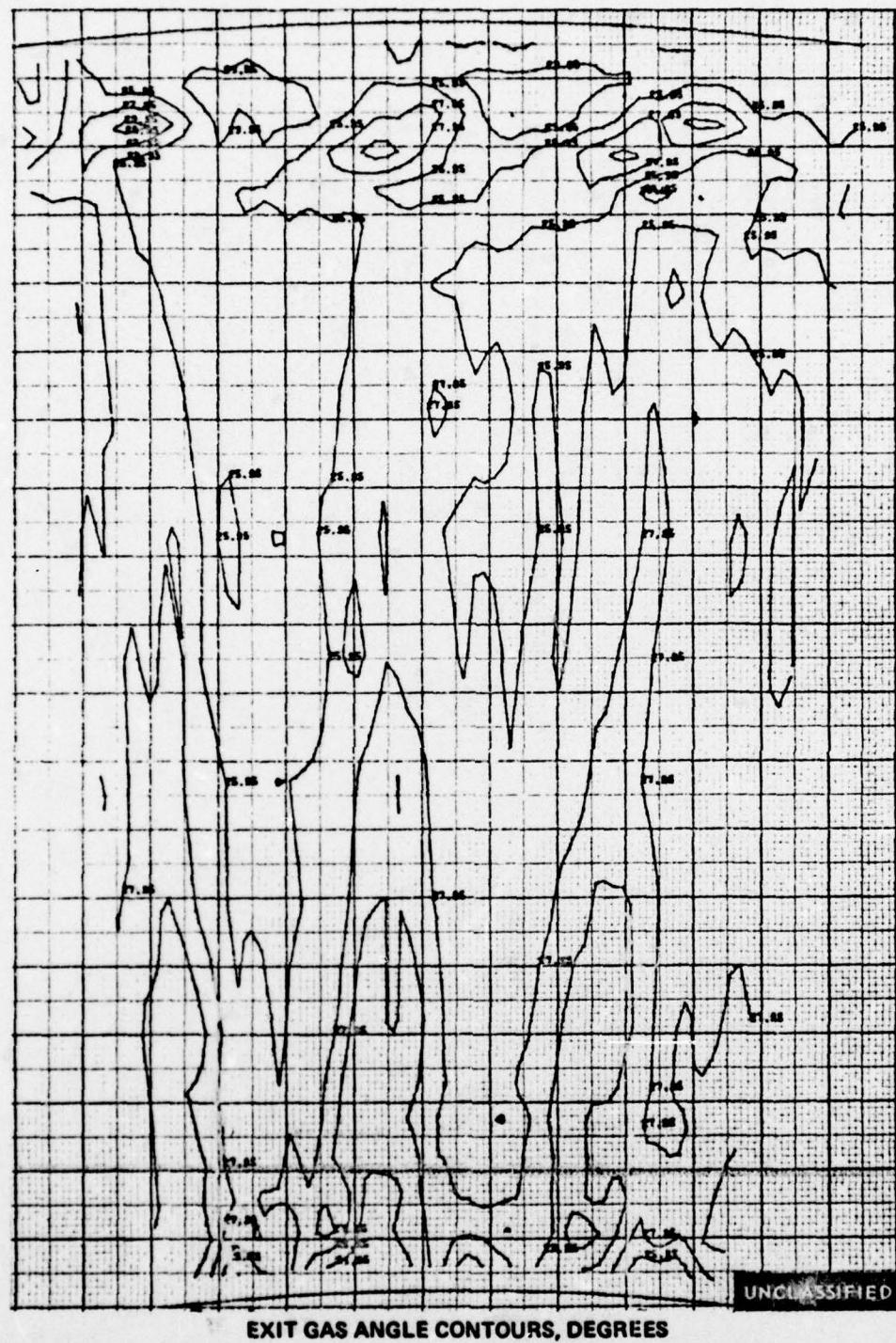
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**Figure 144 Pressure Loss Contours, Second Vane Baseline at -6° Incidence. Three Flow Passages. Midspan Exit Mach Number = 0.865**

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**Figure 145**    Exit Gas Angle Contours, Second Vane Baseline at  $-6^\circ$  Incidence. Three Flow Passages. Midspan Exit Mach Number = 0.865

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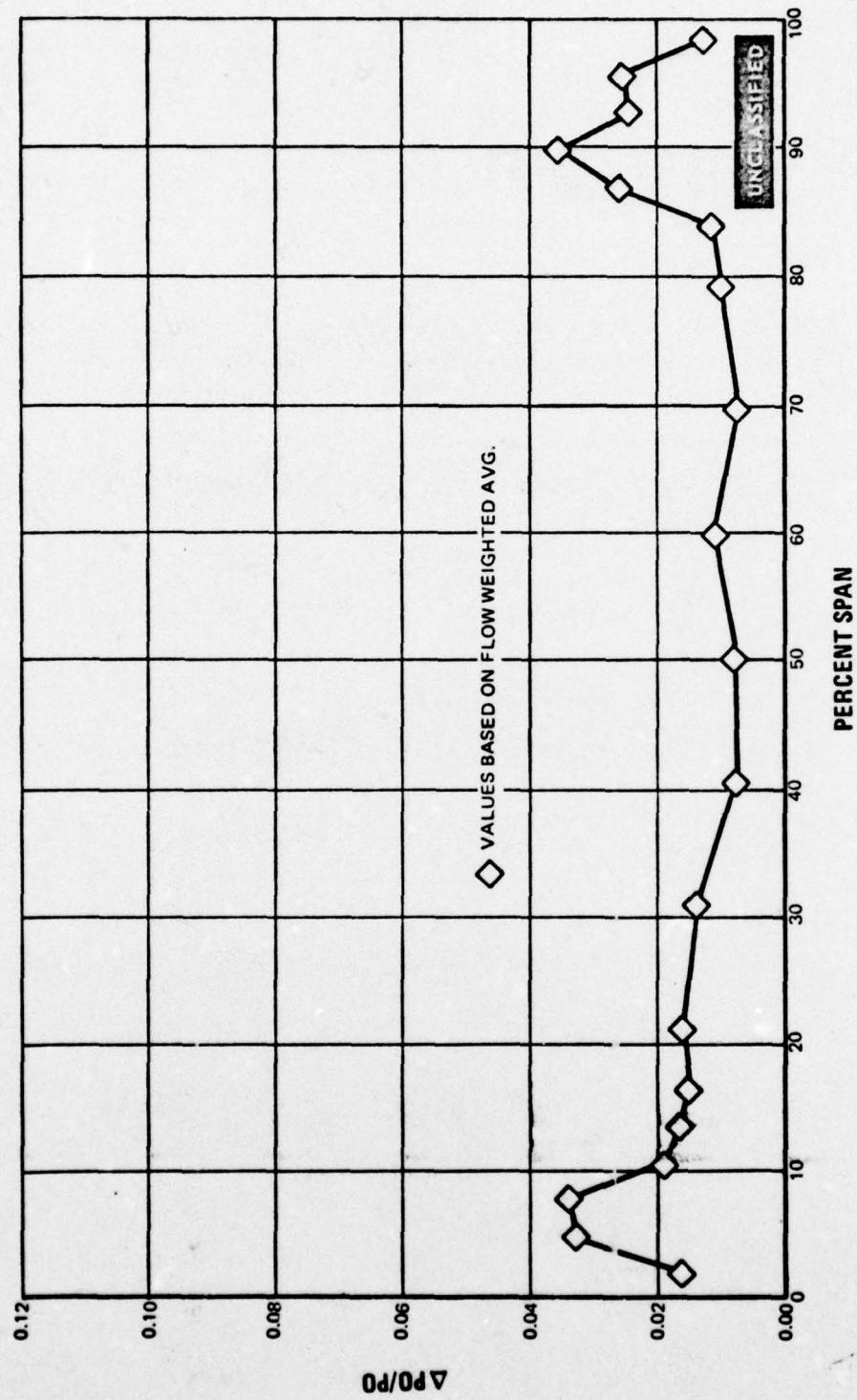


Figure 146 Spanwise Pressure Loss Distribution, Second Vane Baseline at -6° Incidence.  
Midspan Exit Mach Number = 0.865

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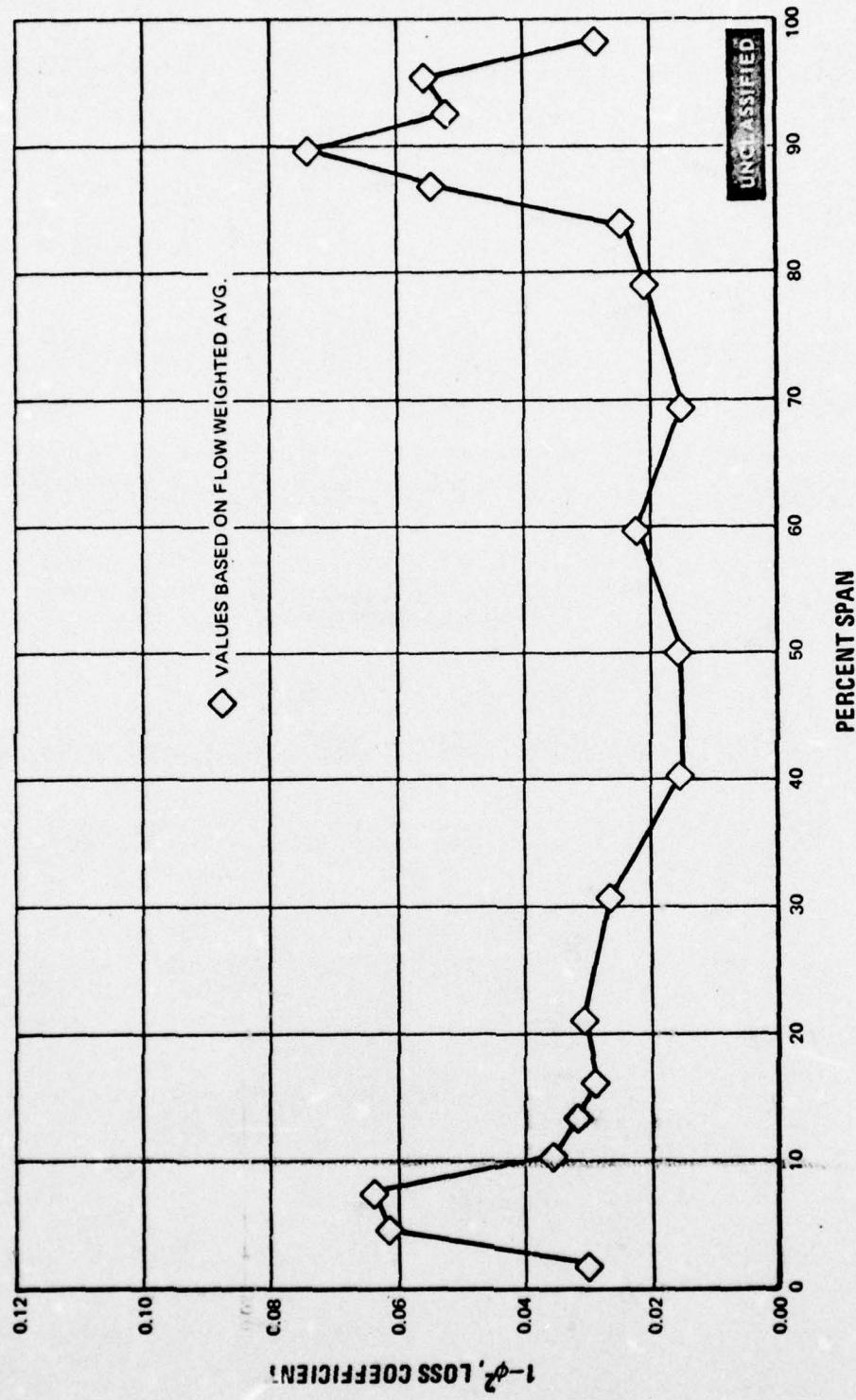
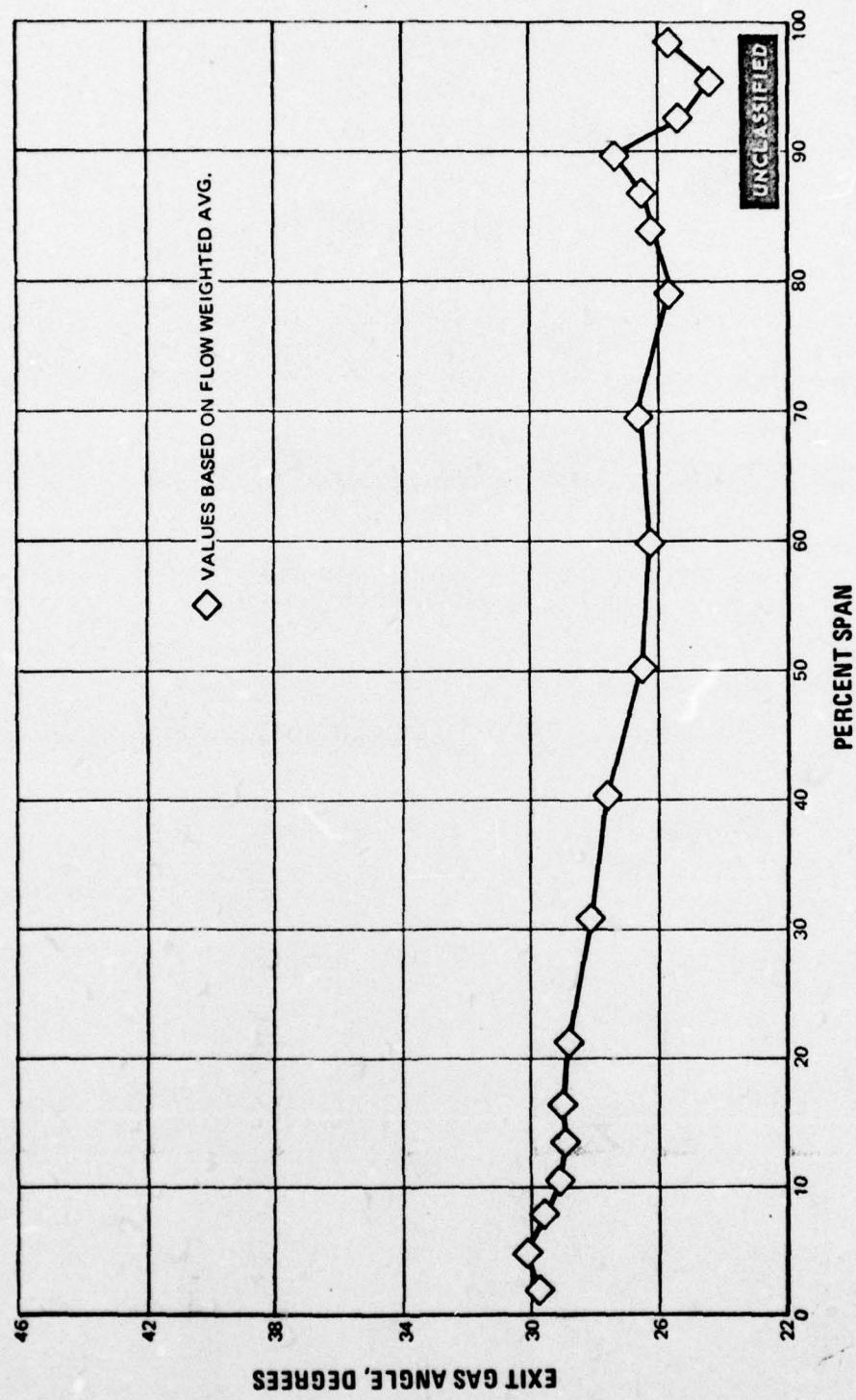


Figure 147 Spanwise Loss Coefficient Distribution, Second Vane Baseline at -6° Incidence. Midspan Exit Mach Number = 0.865

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**Figure 148** Spanwise Exit Gas Angle Distribution, Second Vane Baseline at -6° Incidence.  
Midspan Exit Mach Number = 0.865

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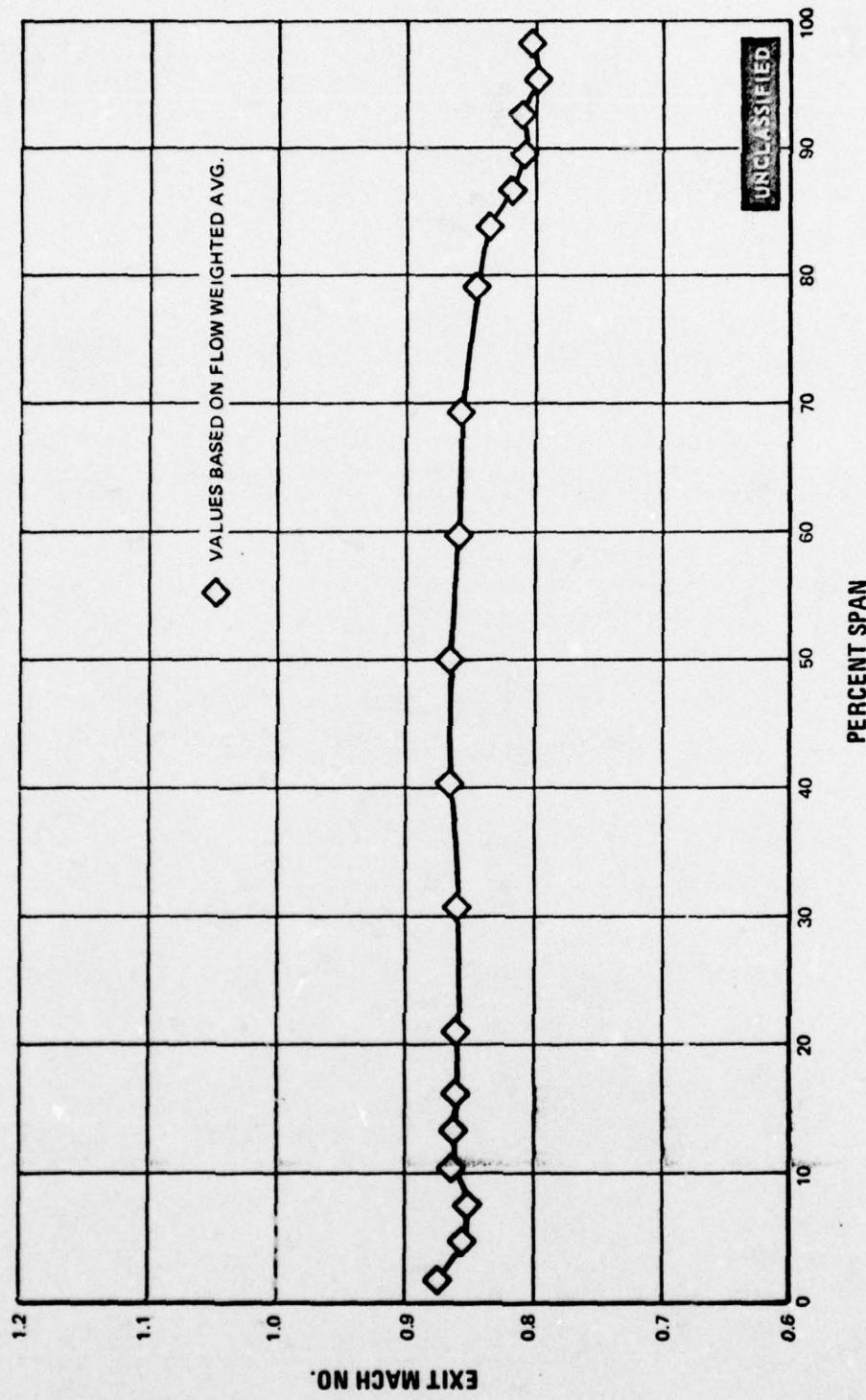


Figure 149 Spanwise Exit Mach Number Distribution, Second Vane Baseline at -6° Incidence. Midspan Exit Mach Number = 0.865

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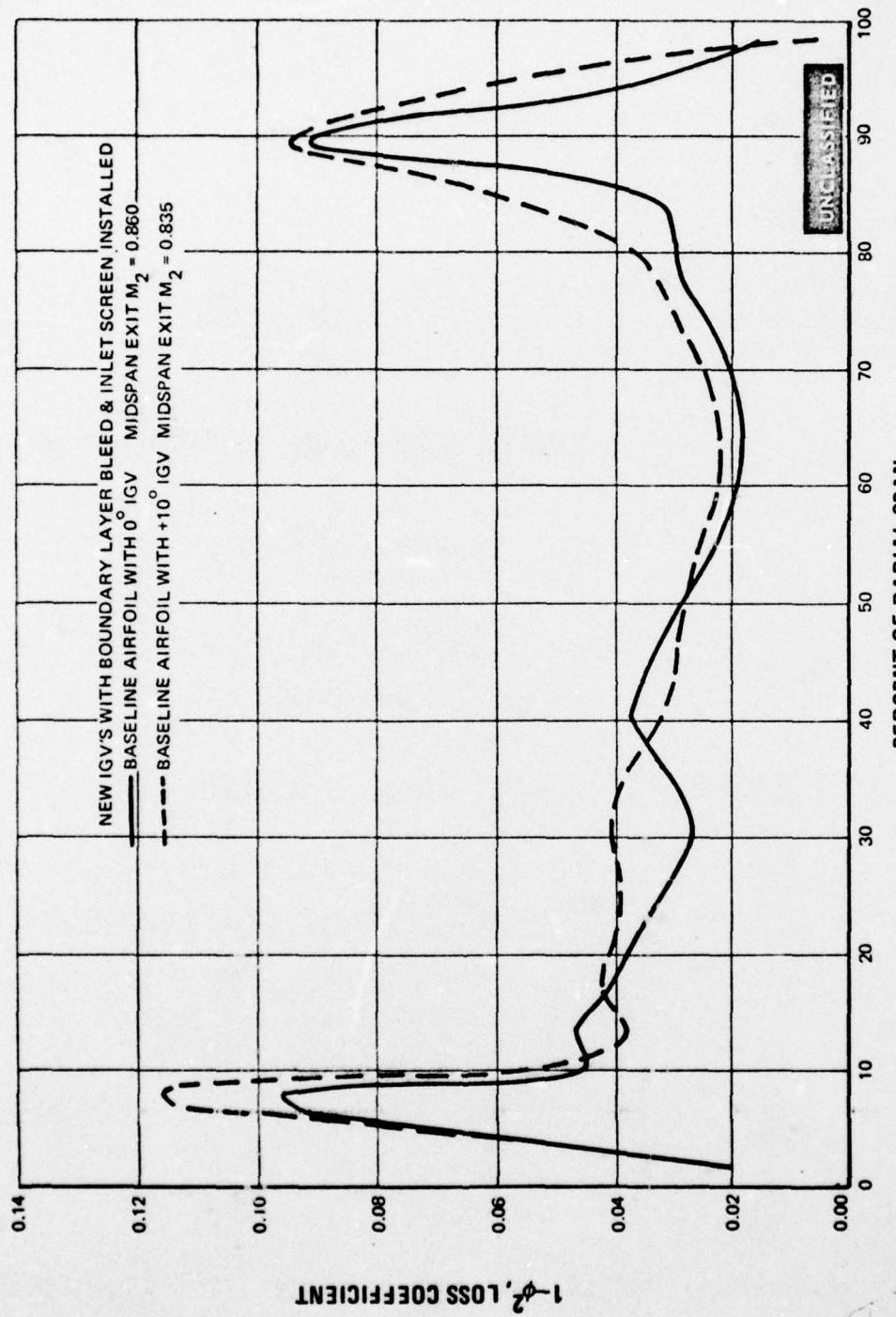


Figure 150 Comparison of Baseline Airfoil Loss Coefficient Distribution at +10° Incidence With Zero Incidence Values

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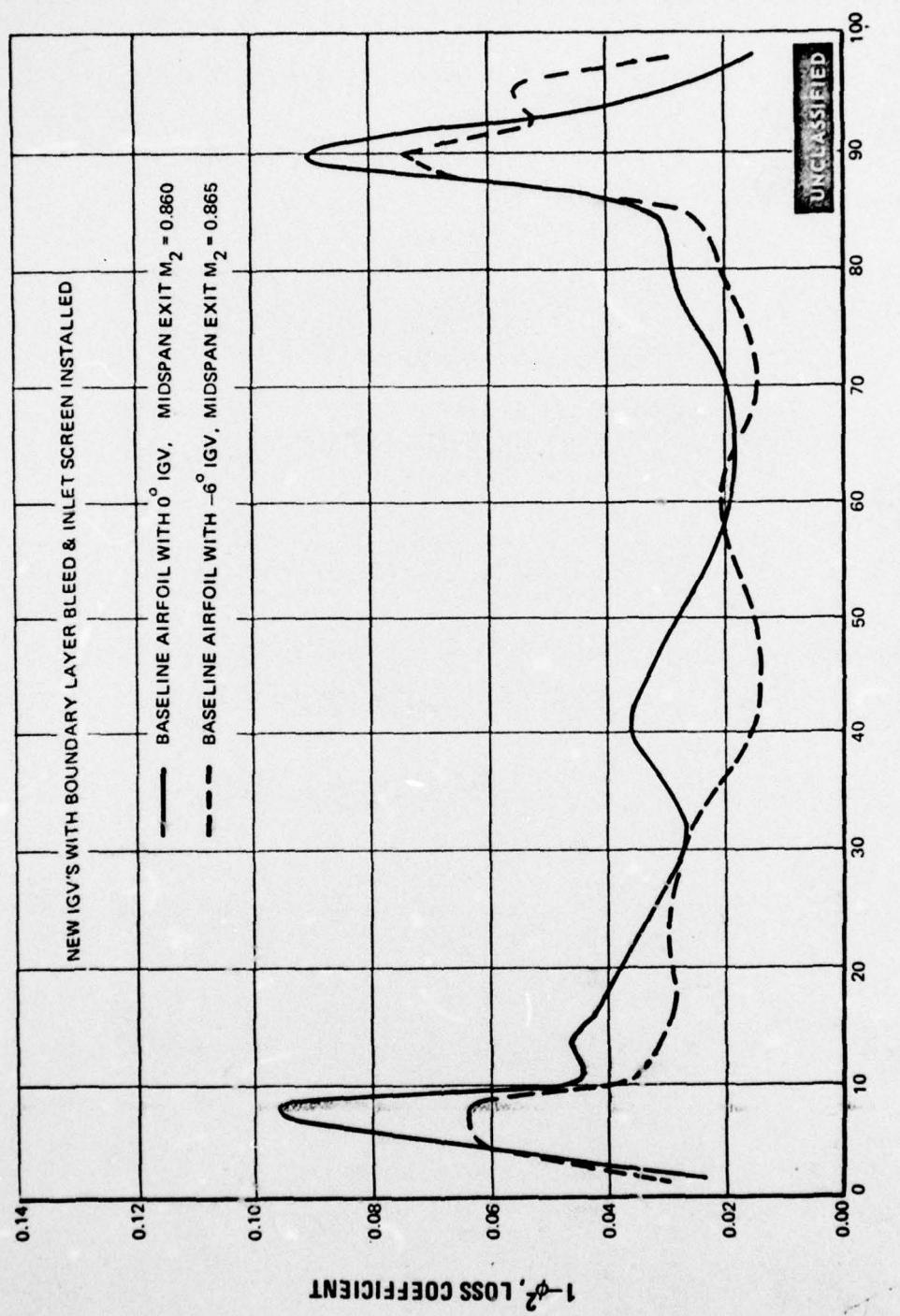


Figure 151 Comparison of Baseline Airfoil Loss Coefficient Distribution at -6 Degree Incidence With Zero Incidence Values

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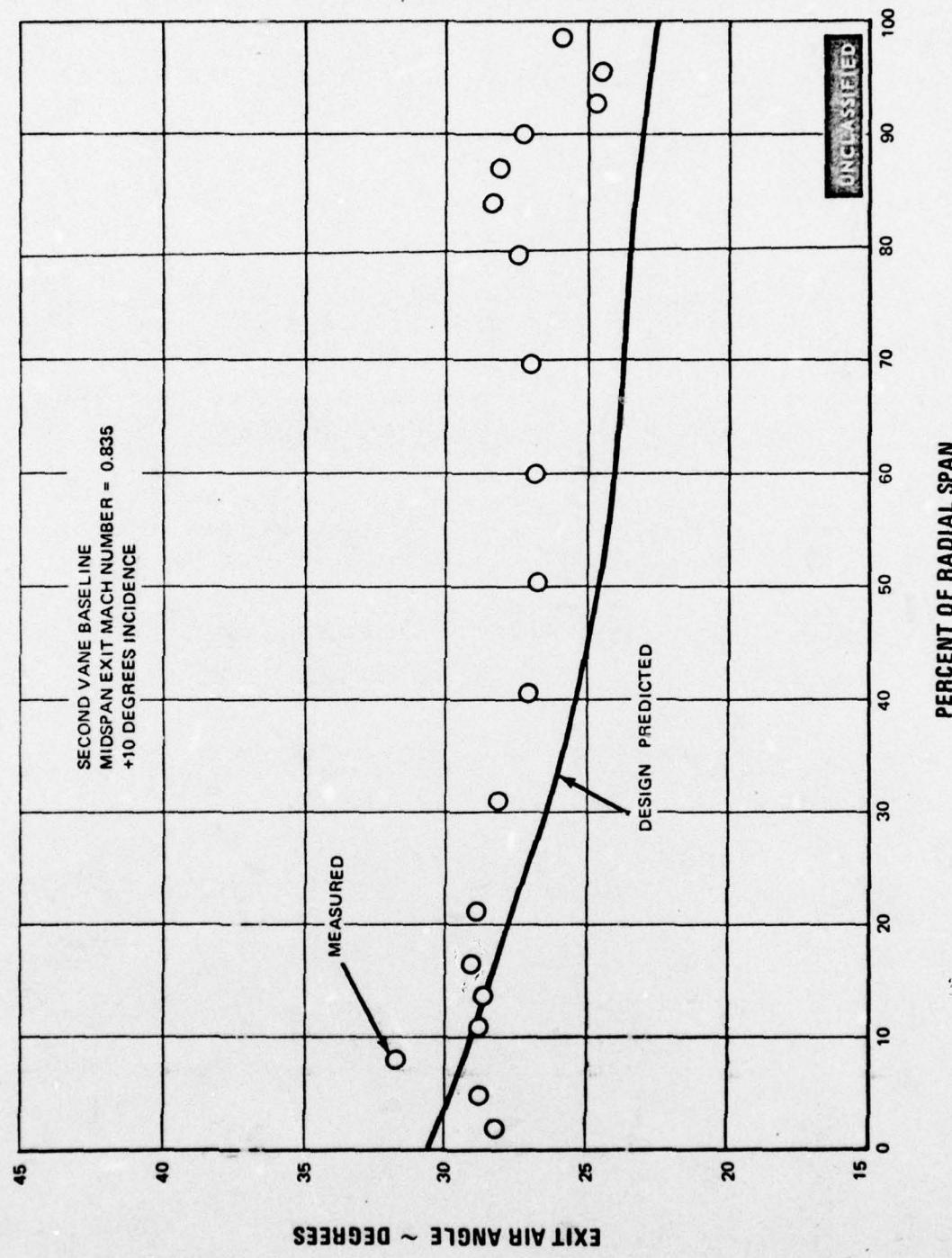


Figure 152 Comparison of Measured Exit Gas Angle With Predicted Design Values at +10 Degrees Incidence. Baseline Airfoil at Midspan Exit Mach No. = 0.835

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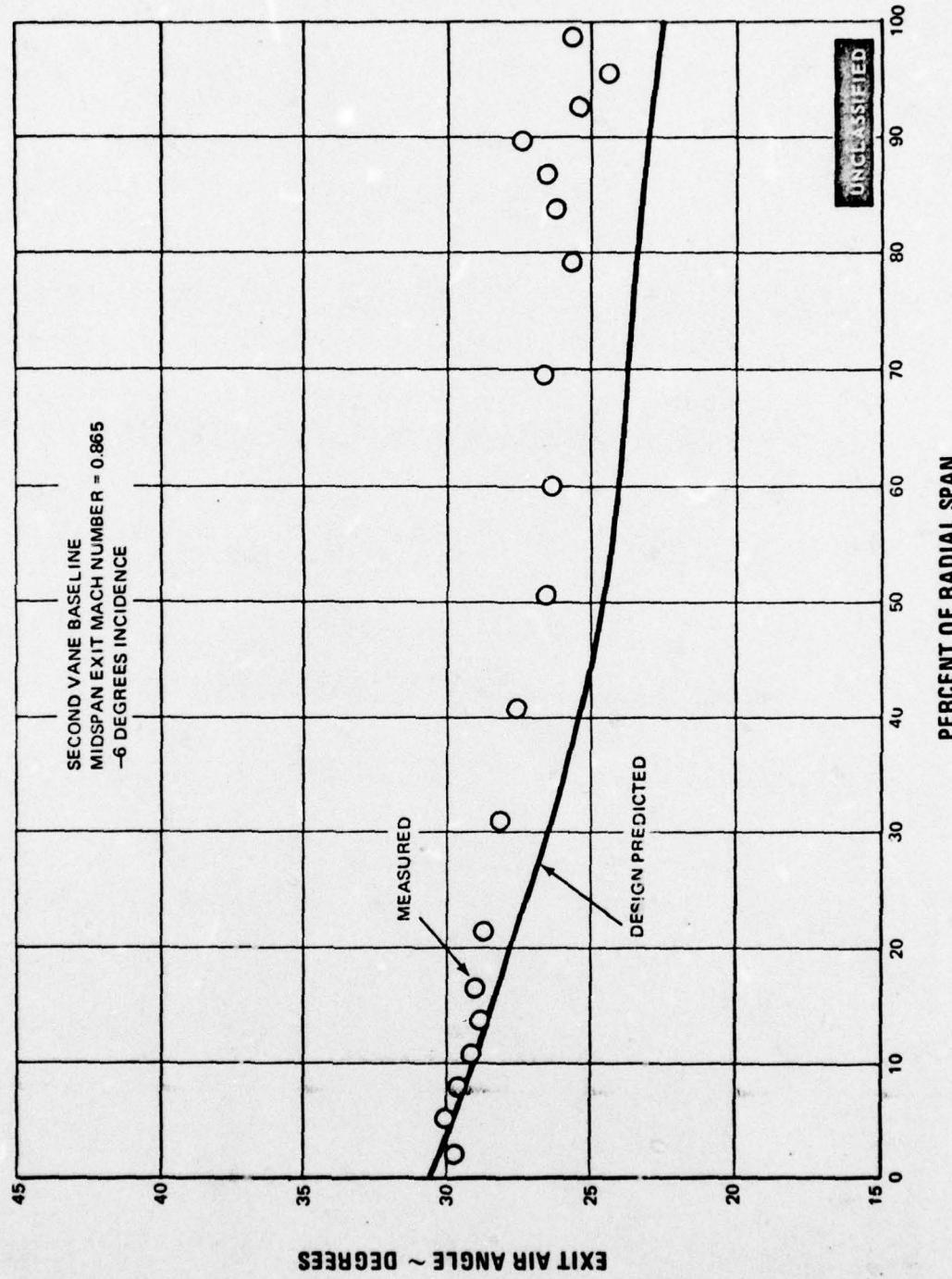


Figure 153 Comparison of Measured Exit Gas Angle With Predicted Design Values at -6 Degrees Incidence. Baseline Airfoil at Midspan Exit Mach No. = 0.865

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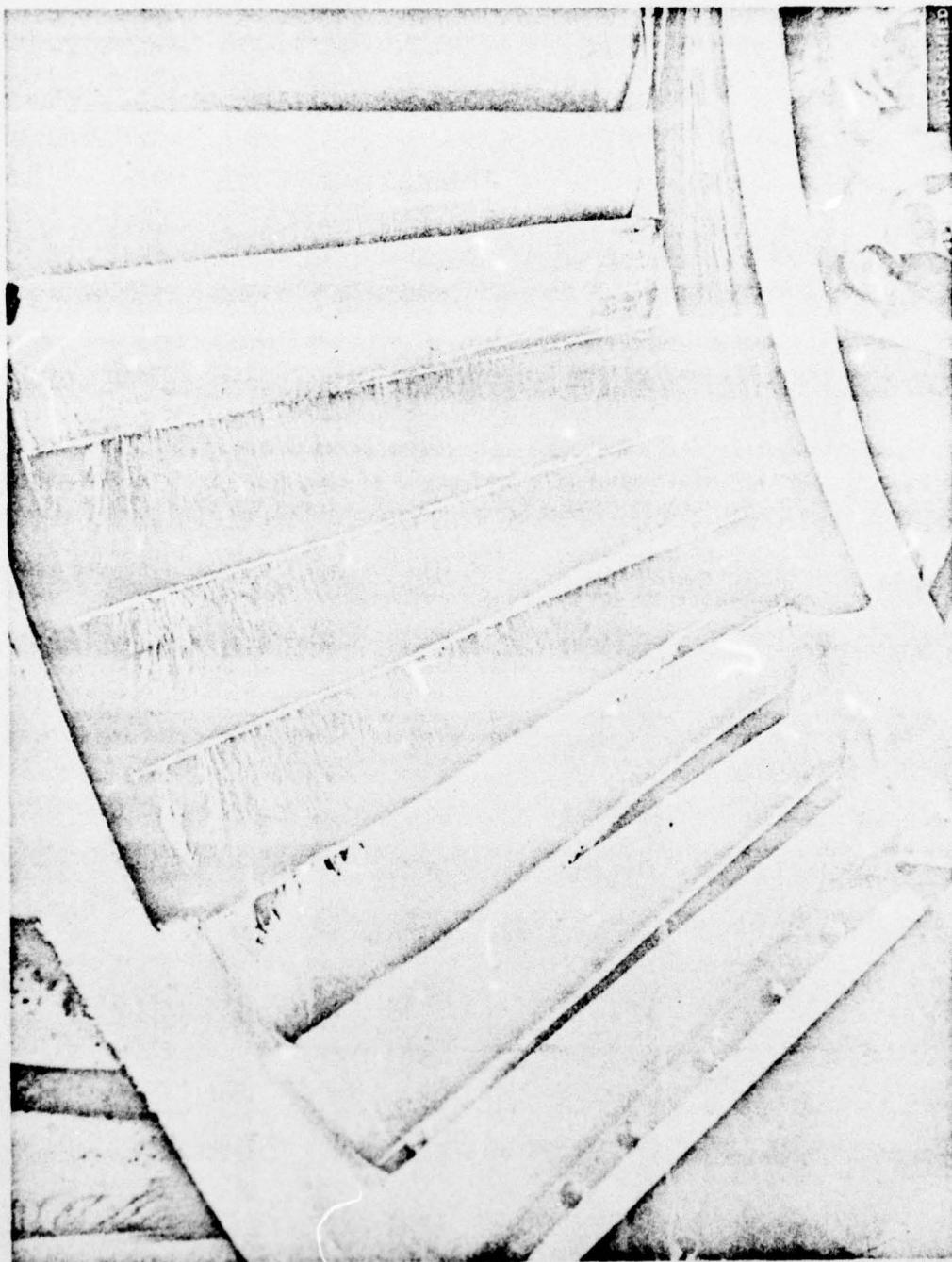


Figure 154 Oil and Graphite Flow Patterns, Second Vane Baseline Airfoil at +10 Degrees Incidence. Midspan Exit Mach No. = 0.835 (FE95349)

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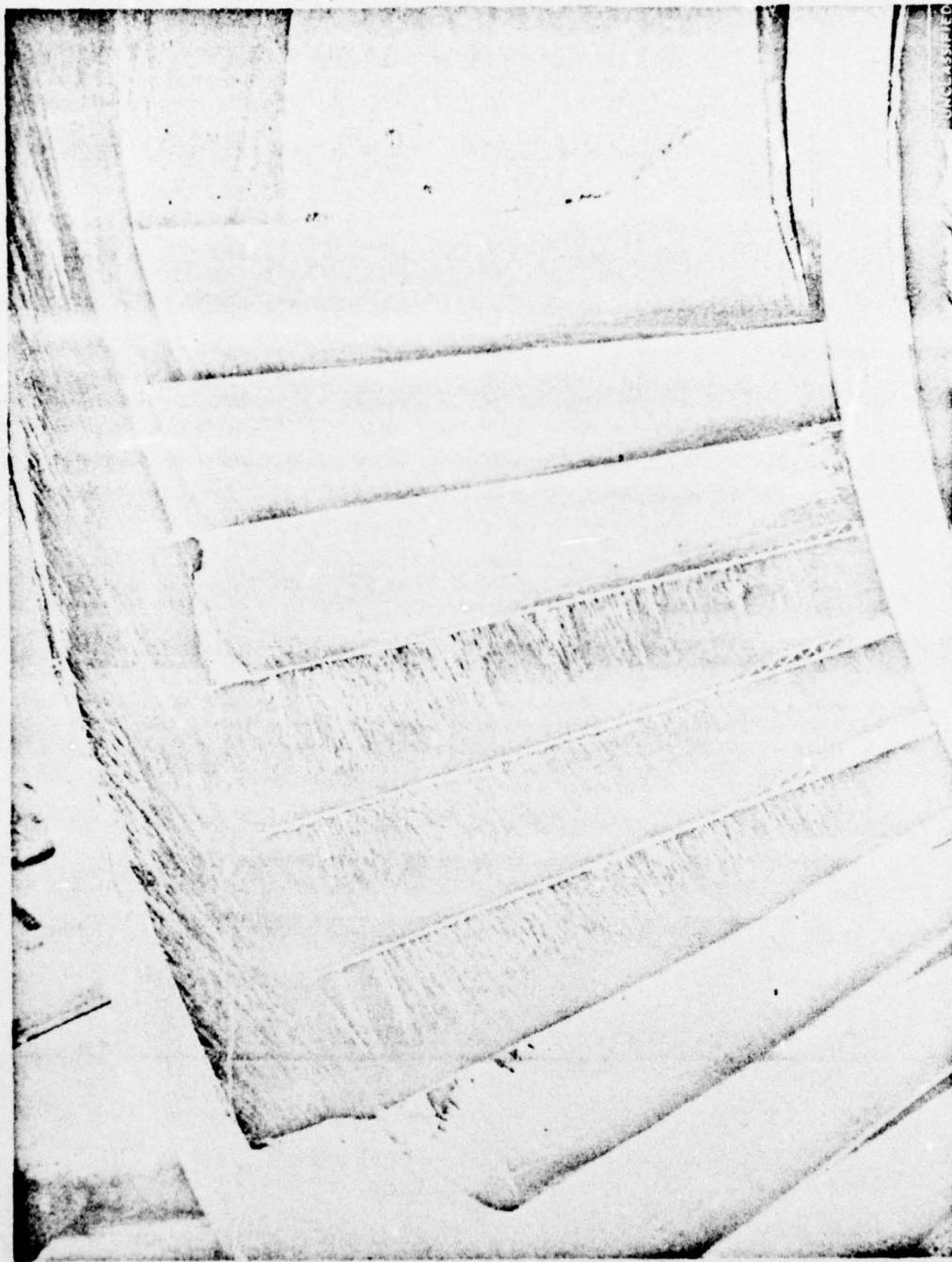


Figure 155 Oil and Graphite Flow Patterns, Second Vane Baseline Airfoil at +10 Degrees Incidence. Midspan Exit Mach No. = 0.835 (FE95350)

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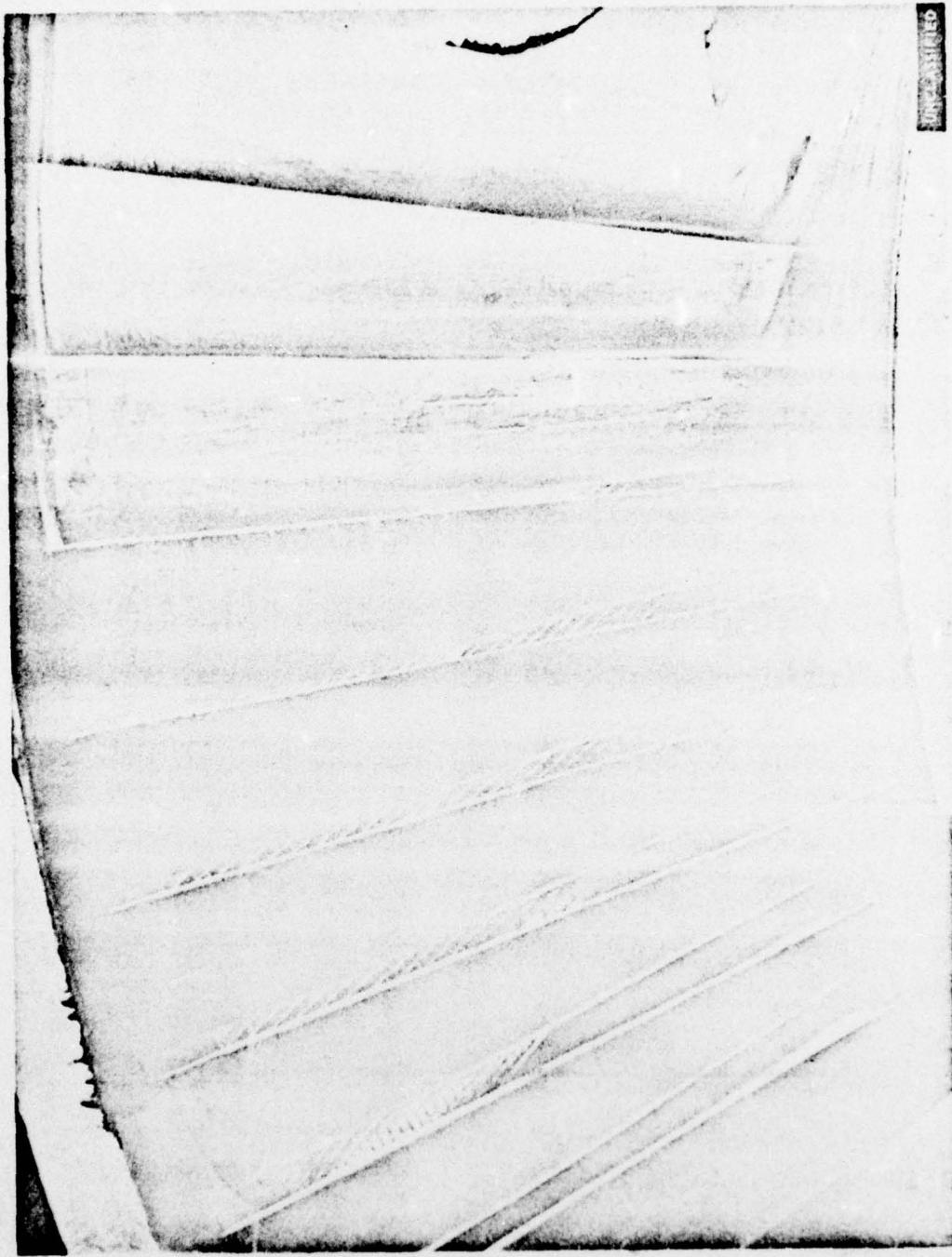
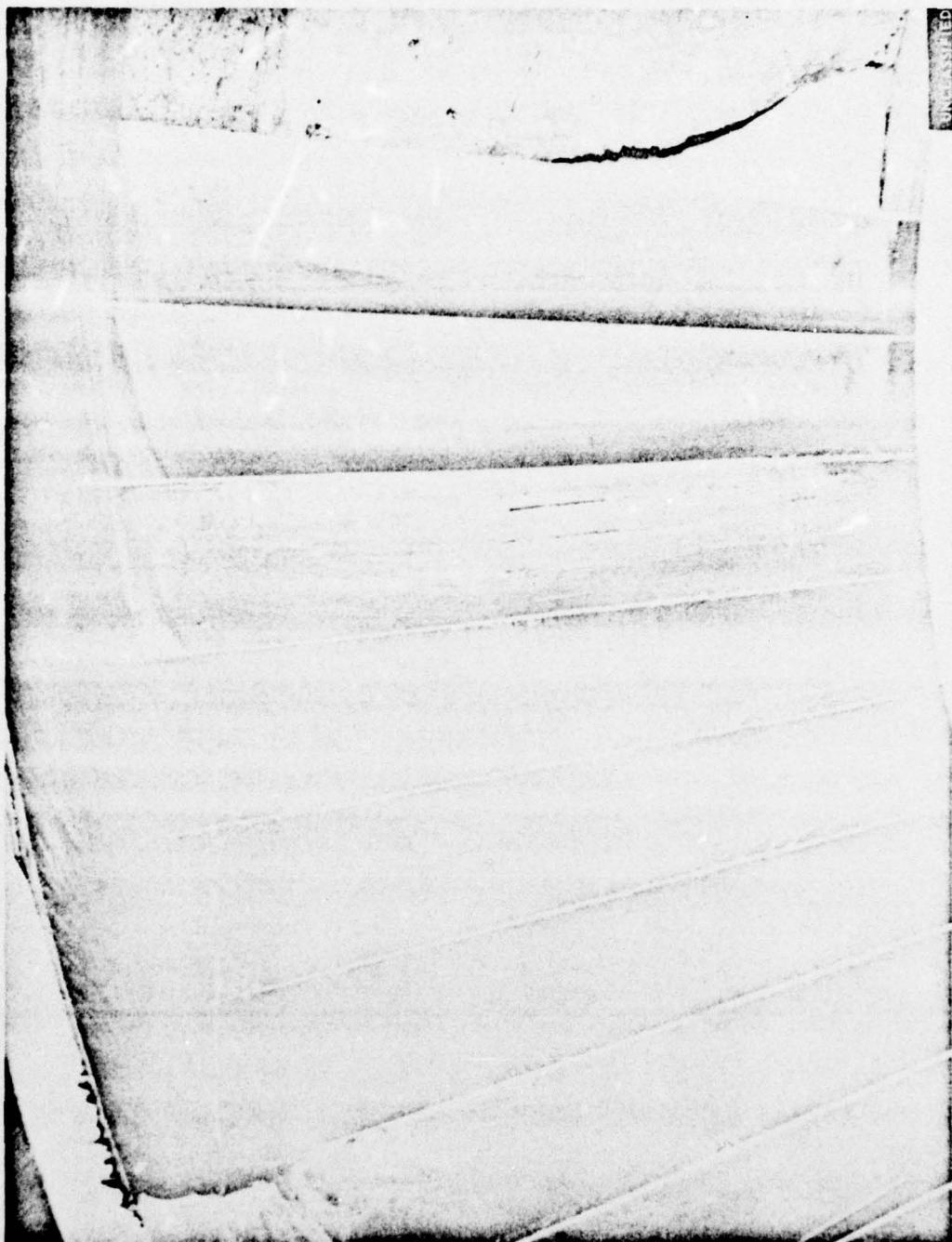


Figure 156 Oil and Graphite Flow Patterns, Second Vane Baseline Airfoil at -6 Degrees Incidence. Midspan Exit Mach No. = 0.865 (FE95432)

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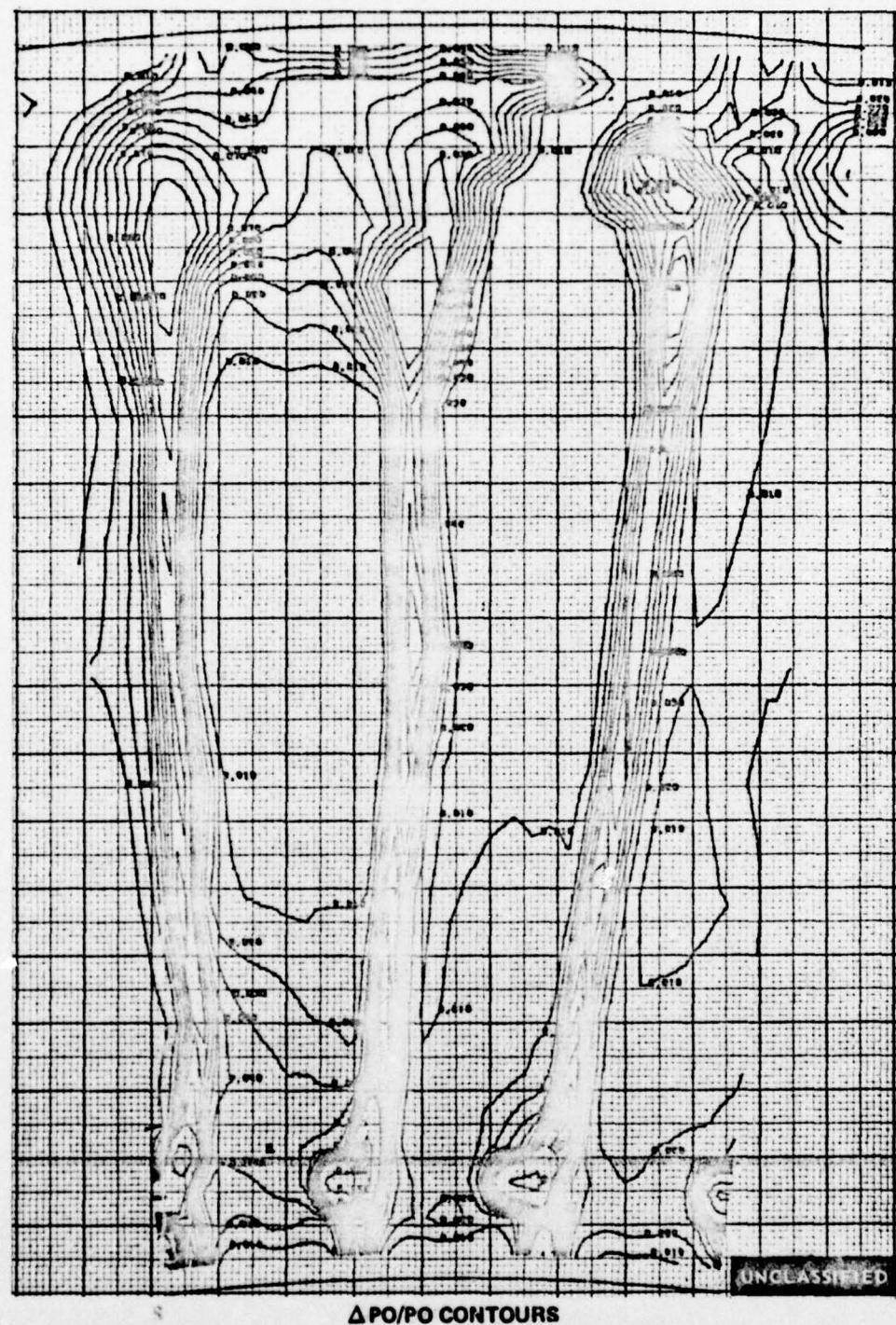
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**Figure 157 Oil and Graphite Flow Patterns, Second Vane Baseline Airfoil at -6 Degrees Incidence. Midspan Exit Mach No. = 0.865 (FE95433)**

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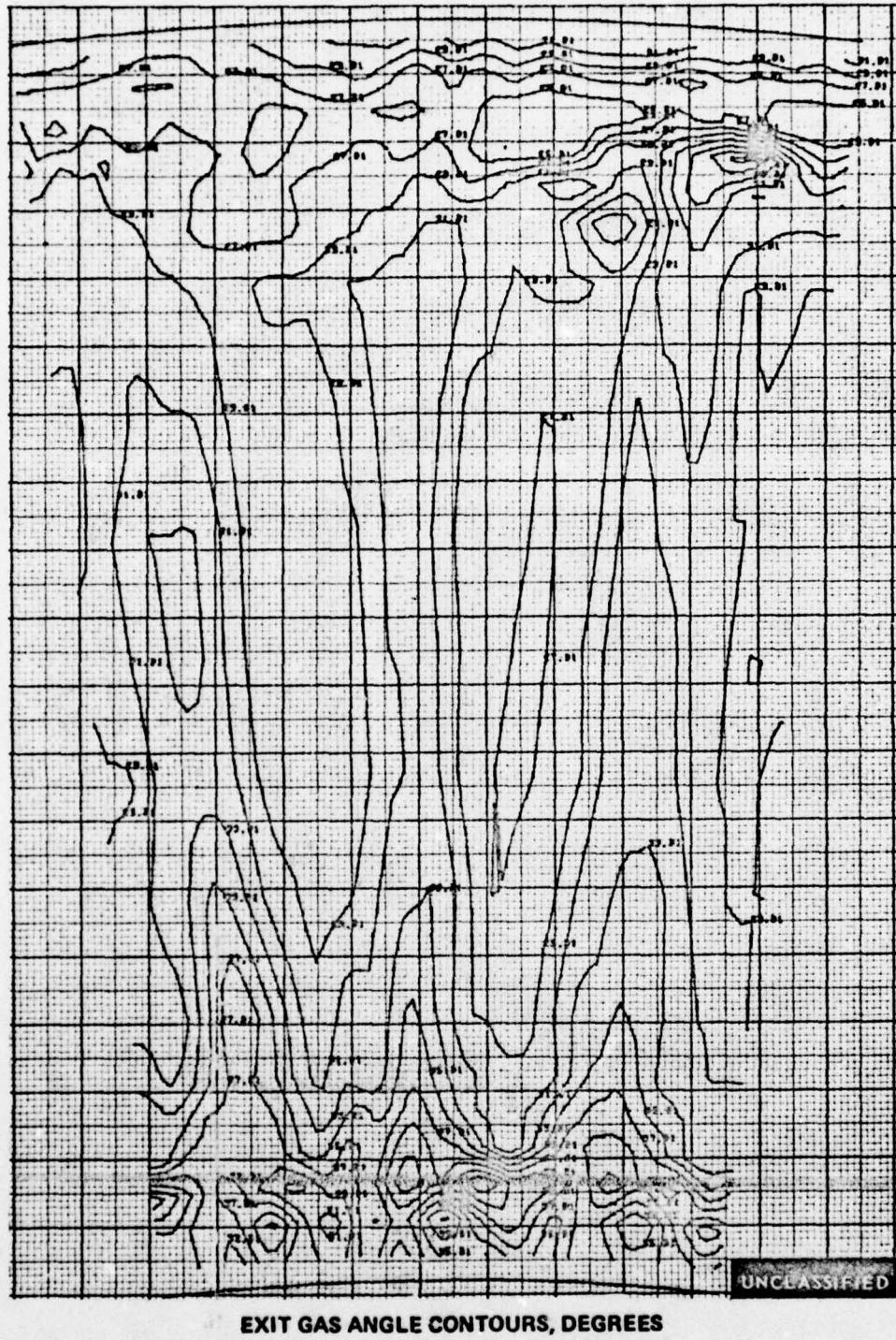
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**Figure 158 Pressure Loss Contours, Second Vane First Recambering at  $+10^\circ$  Incidence.  
Three Flow Passages. Midspan Exit Mach Number = 0.840**

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**Figure 159 Exit Gas Angle Contours, Second Vane First Recambering at +10° Incidence. Three Flow Passages. Midspan Exit Mach Number = 0.840**

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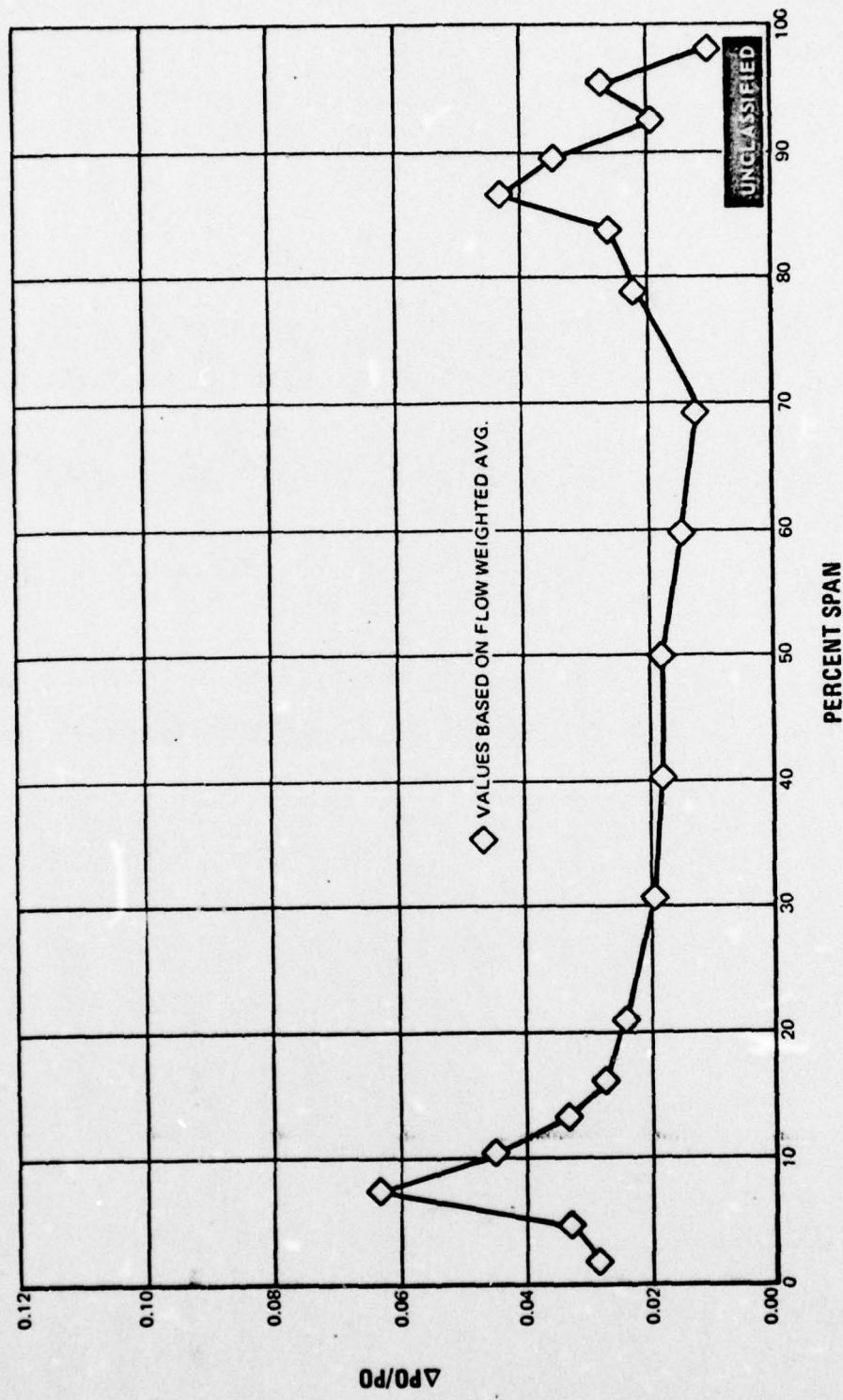


Figure 160 Spanwise Pressure Loss Distribution, Second Vane First Recambering at  
+10° Incidence. Midspan Exit Mach Number = 0.840

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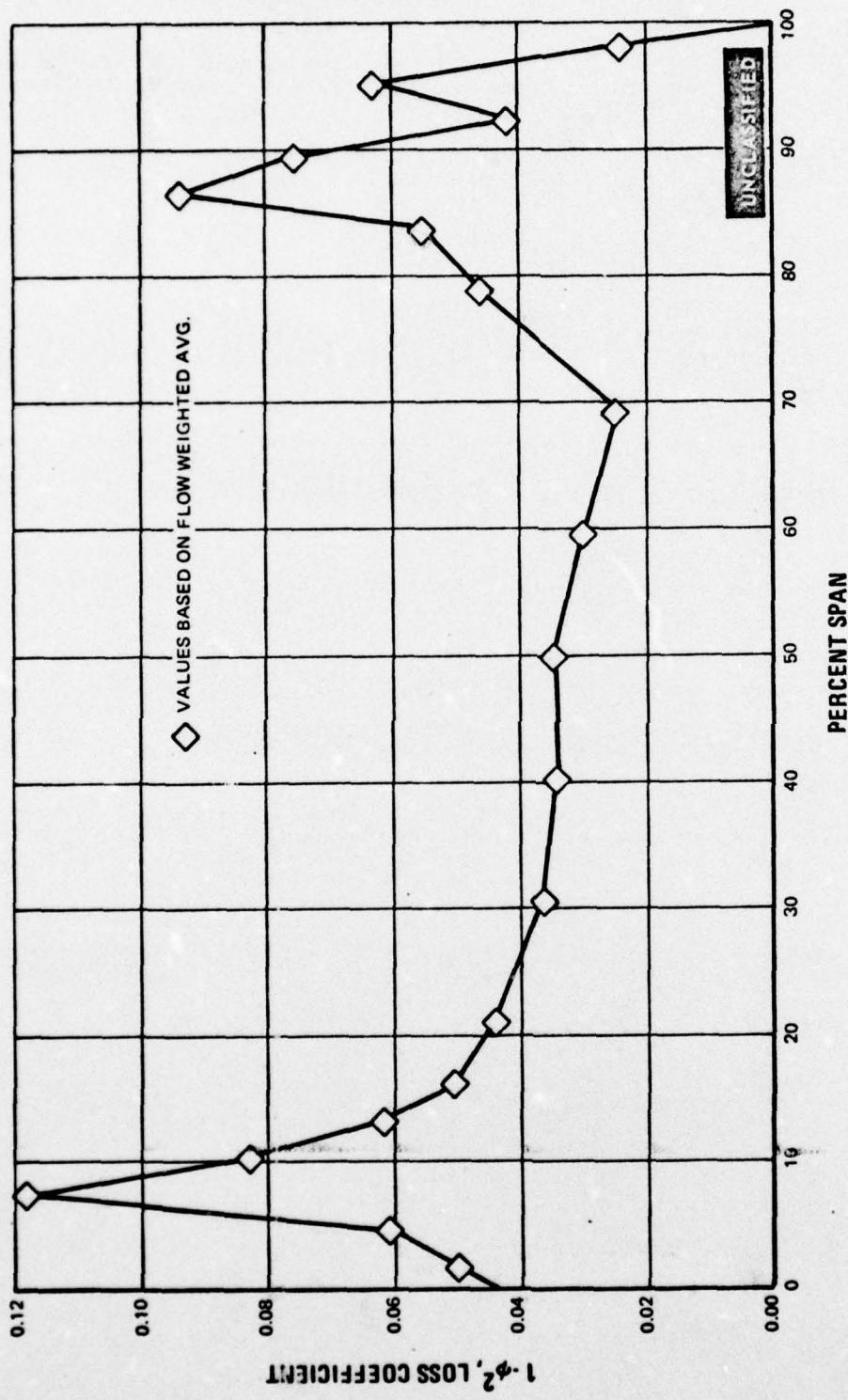


Figure 161 Spanwise Loss Coefficient Distribution, Second Vane First Recambering at  
+10° Incidence. Midspan Exit Mach Number = 0.840

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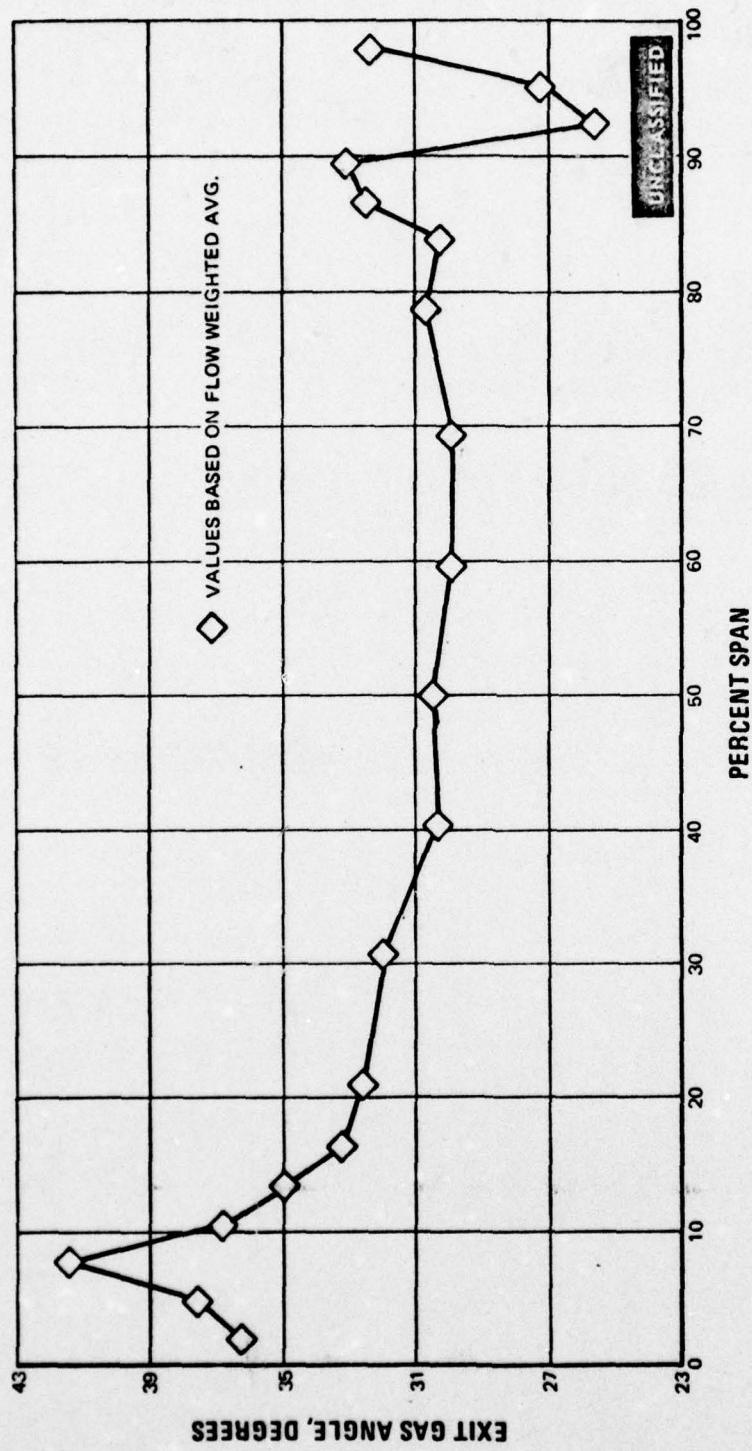


Figure 162 Spanwise Exit Gas Angle Distribution, Second Vane First Recambering at +10° Incidence. Midspan Exit Mach Number = 0.840

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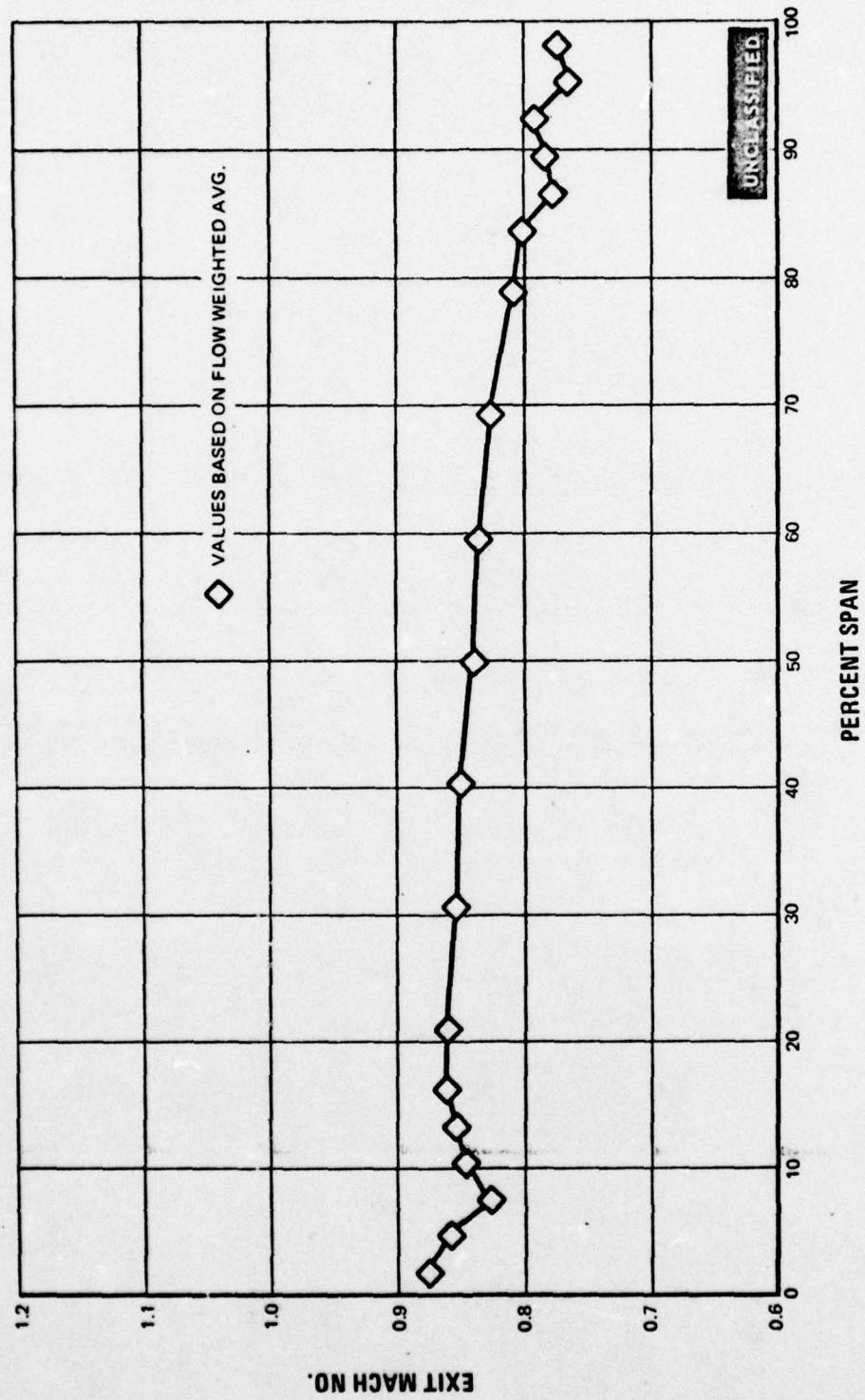
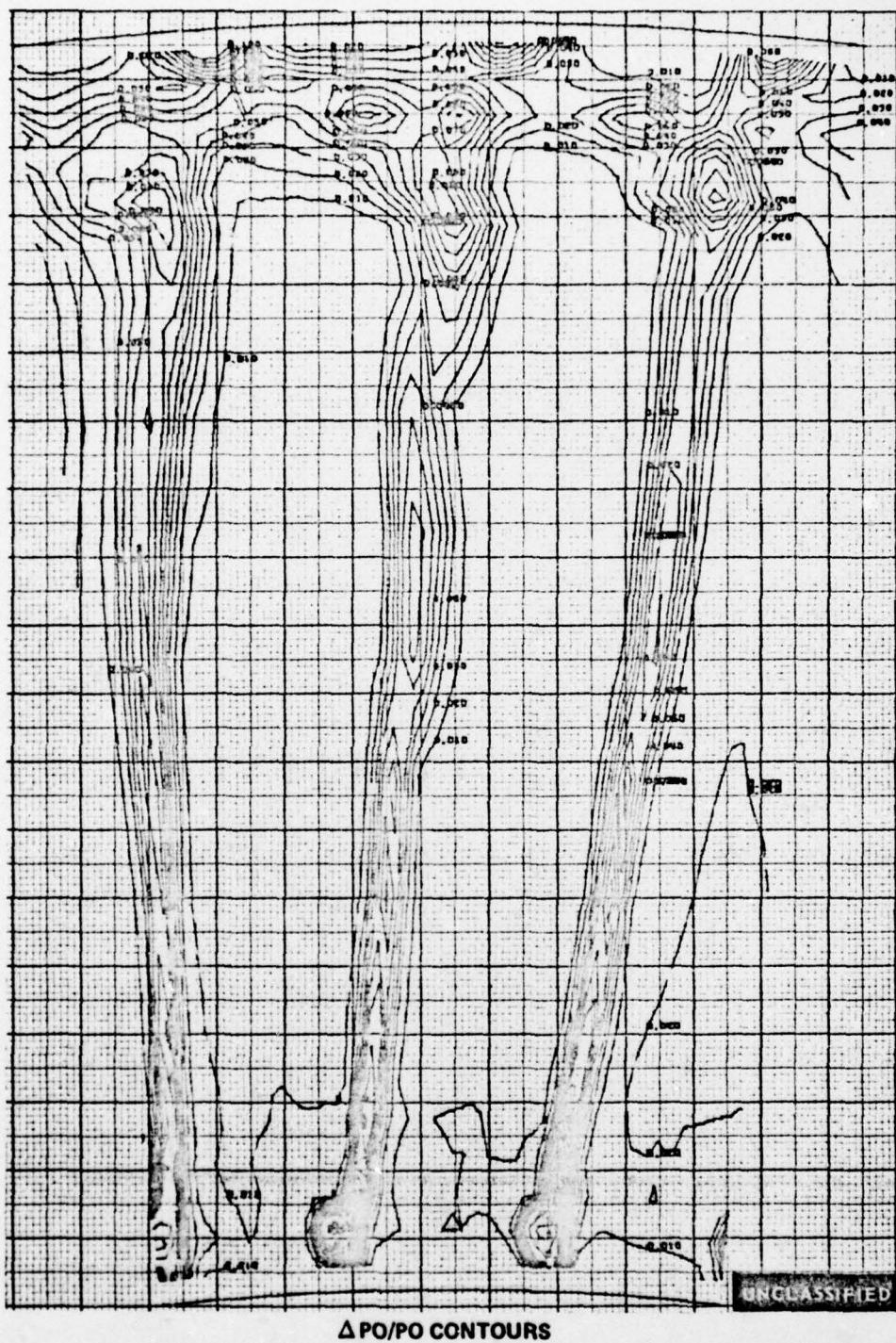


Figure 163 Spanwise Exit Mach Number Distribution, Second Vane First Recambering  
at  $+10^\circ$  Incidence. Midspan Exit Mach Number = 0.840

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**Figure 164** Pressure Loss Contours, Second Vane First Recambering at  $-6^\circ$  Incidence. Three Flow Passages. Midspan Exit Mach Number = 0.883

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**Figure 165** Exit Gas Angle Contours, Second Vane First Recombining at -6° Incidence.  
Three Flow Passages. Midspan Exit Mach Number = 0.883

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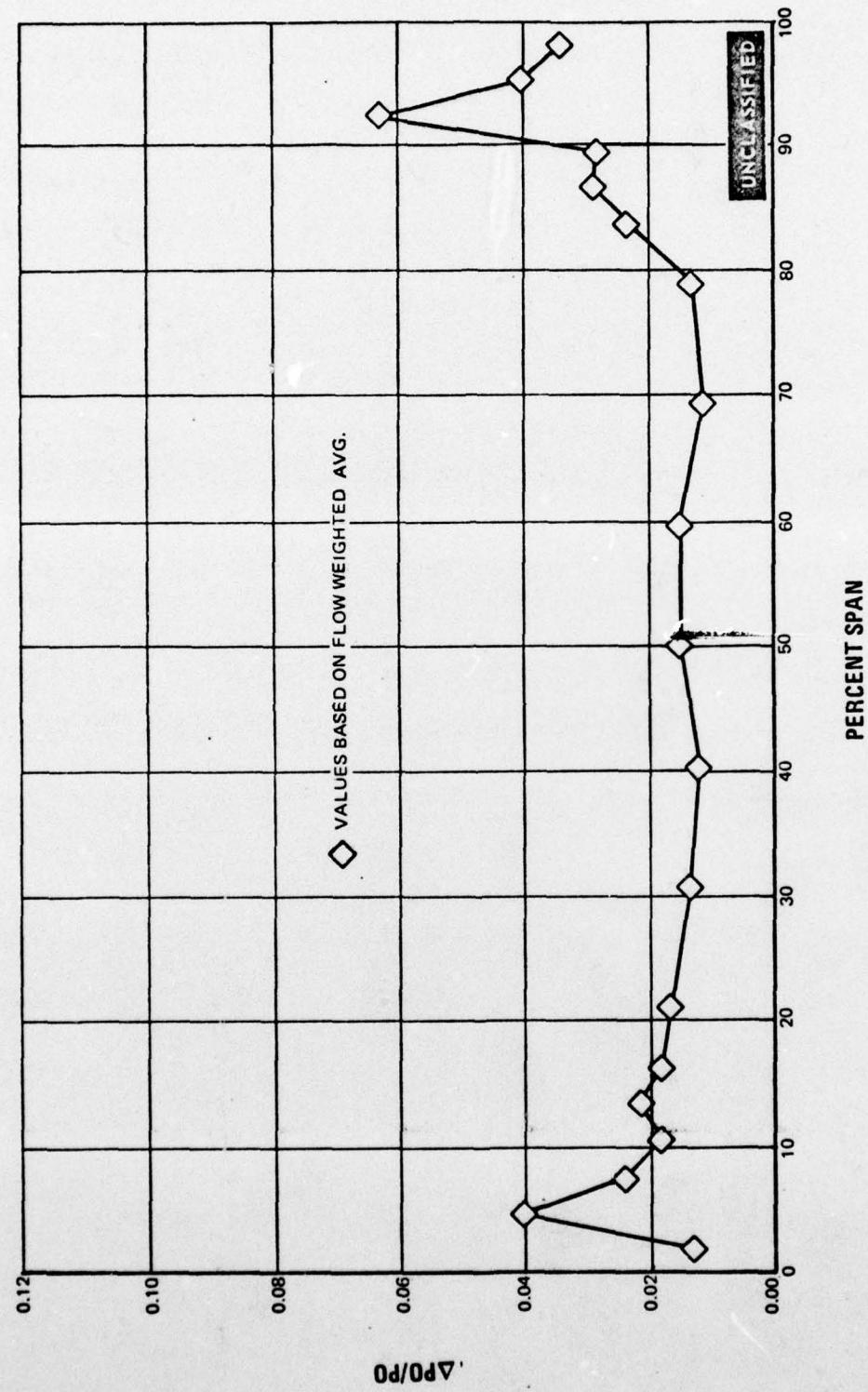


Figure 166 Spanwise Pressure Loss Distribution, Second Vane First Recambering at -6° Incidence. Midspan Exit Mach Number = 0.883

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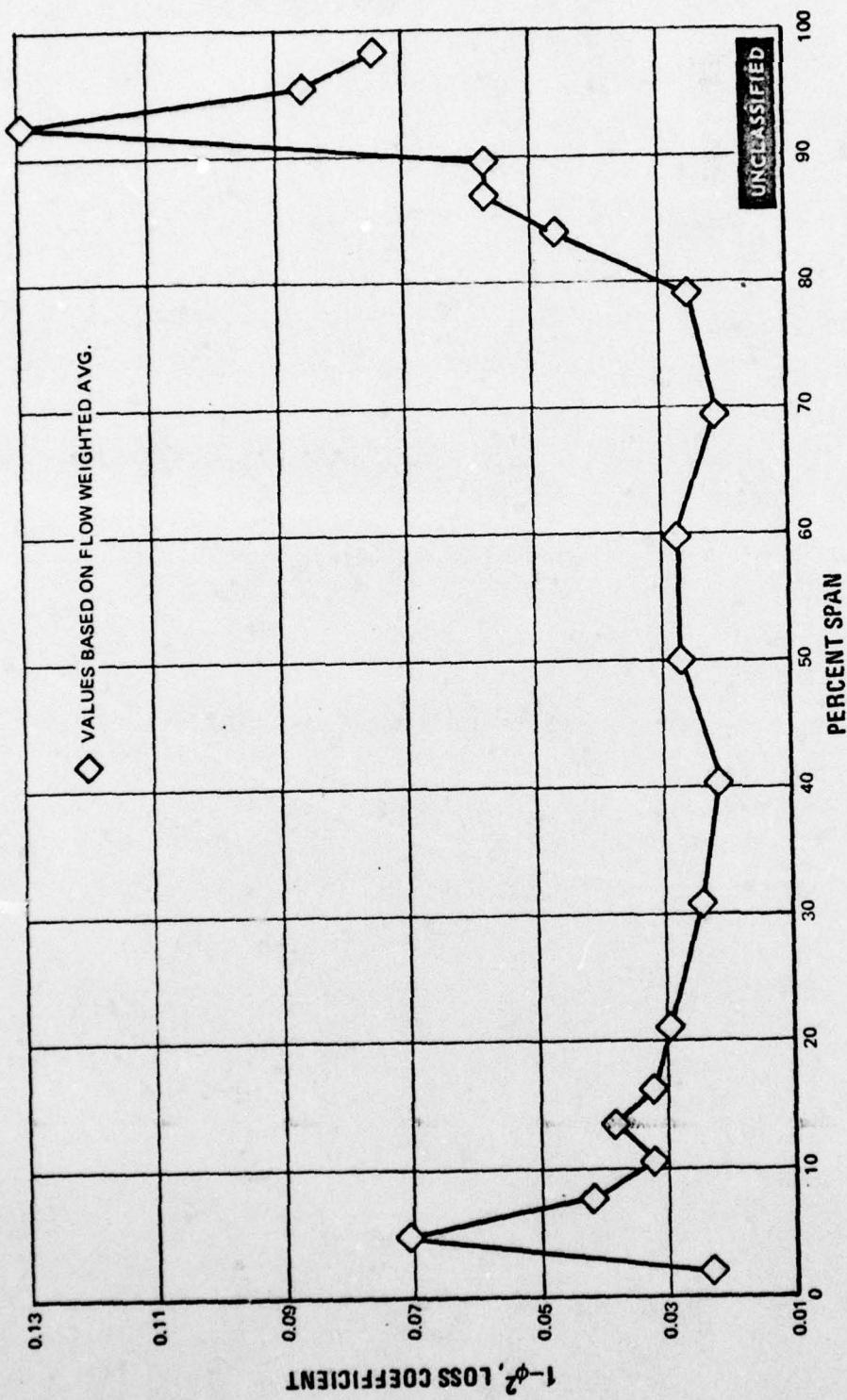


Figure 167 Spanwise Loss Coefficient Distribution, Second Vane First Recambering at  
-6° Incidence. Midspan Exit Mach Number = 0.883

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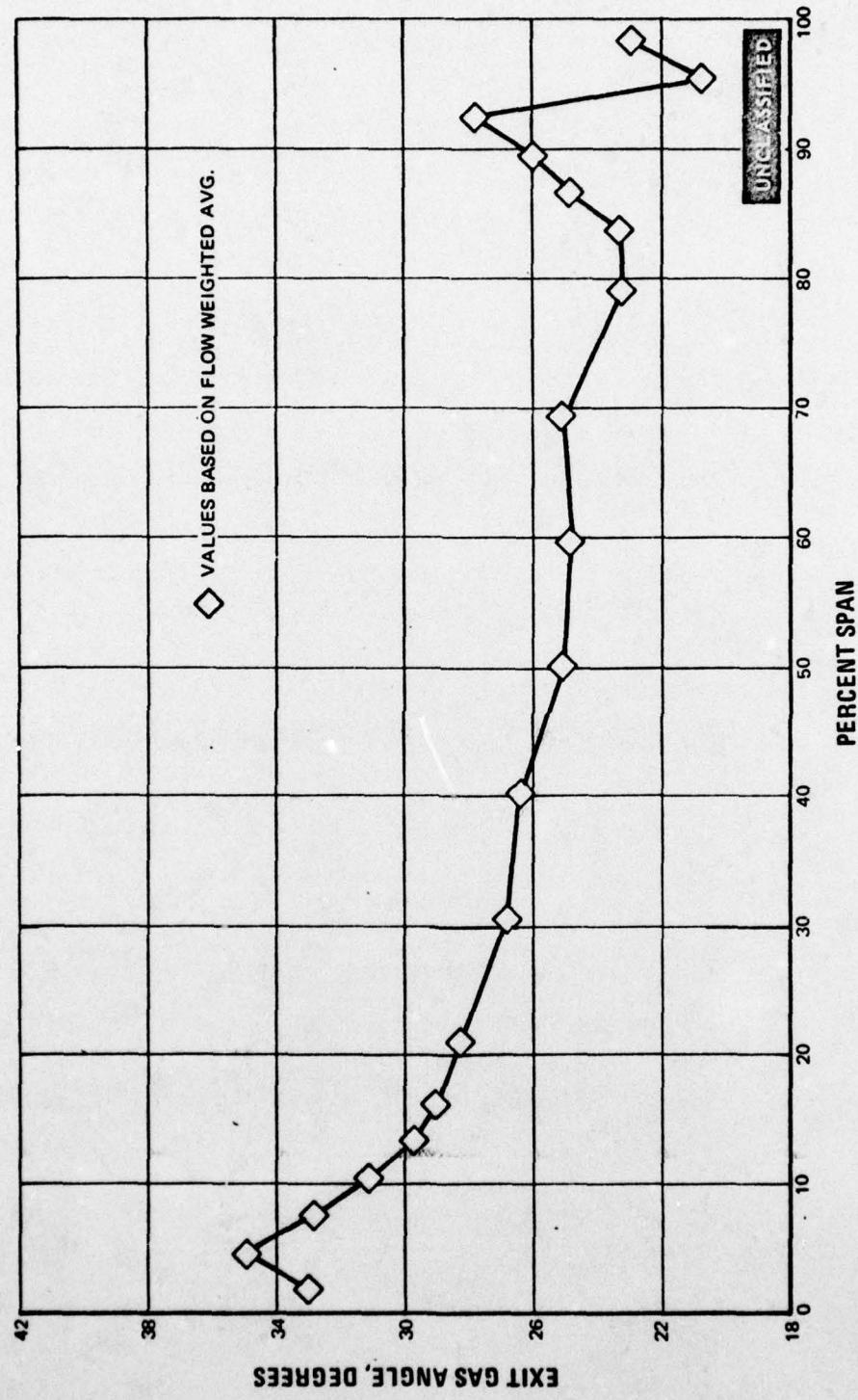


Figure 168 Spanwise Exit Gas Angle Distribution, Second Vane First Recombining at  
-6° Incidence. Midspan Exit Mach Number = 0.883

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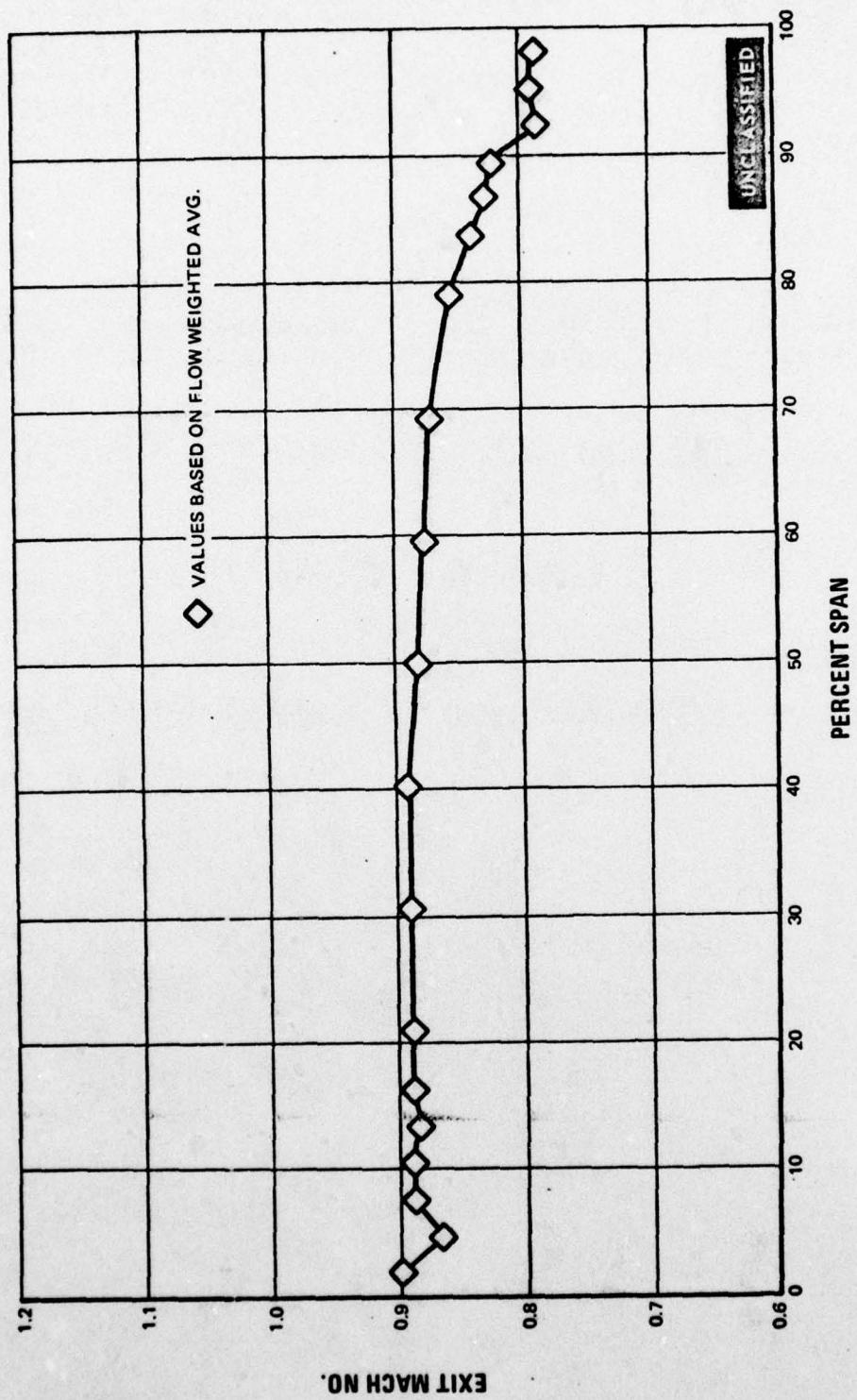


Figure 169 Spanwise Exit Mach Number Distribution, Second Vane First Recambering  
at -6° Incidence. Midspan Exit Mach Number = 0.883

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(U) Comparisons of the spanwise loss coefficient distribution at +10 degrees incidence with zero degrees incidence, and -6 degrees incidence with zero degrees incidence are shown in Figures 170 and 171, respectively. The root end-wall losses were larger for the positive incidence and almost identical for the negative incidence values when compared to the zero incidence results. The tip end-wall losses showed very little variation for the three incidences that were investigated. On an overall basis, the positive incidence loss coefficient was substantially larger, and the negative incidence value was slightly greater than for the zero incidence condition. The measured exit gas angles are shown only for the -6 degree condition (Figure 172), since probe problems during the +10 degree test produced erroneous angle readings. The measured angles, as was the case for the baseline tests, were in good agreement with the design predicted values. Oil and graphite flow patterns are shown in Figures 173 through 176 for both positive and negative incidence tests. The flow was, as usual, well-behaved.

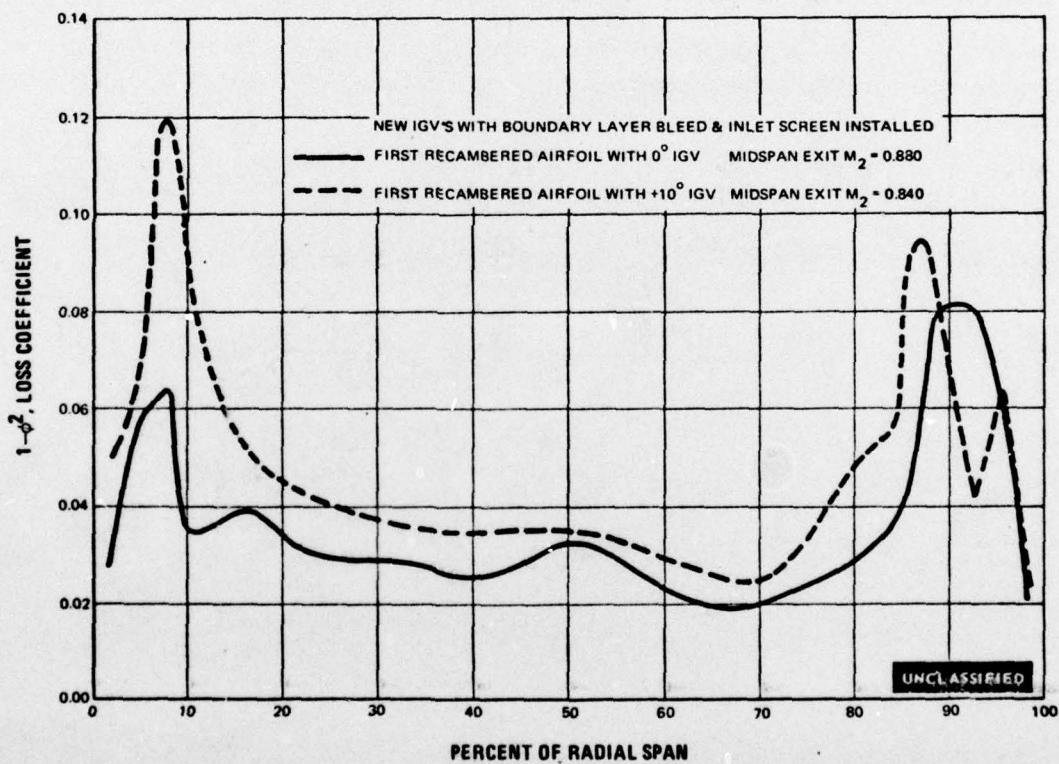


Figure 170 Comparison of First Recambered Airfoil Loss Coefficient Distribution at +10 Degrees Incidence With Zero Incidence Values

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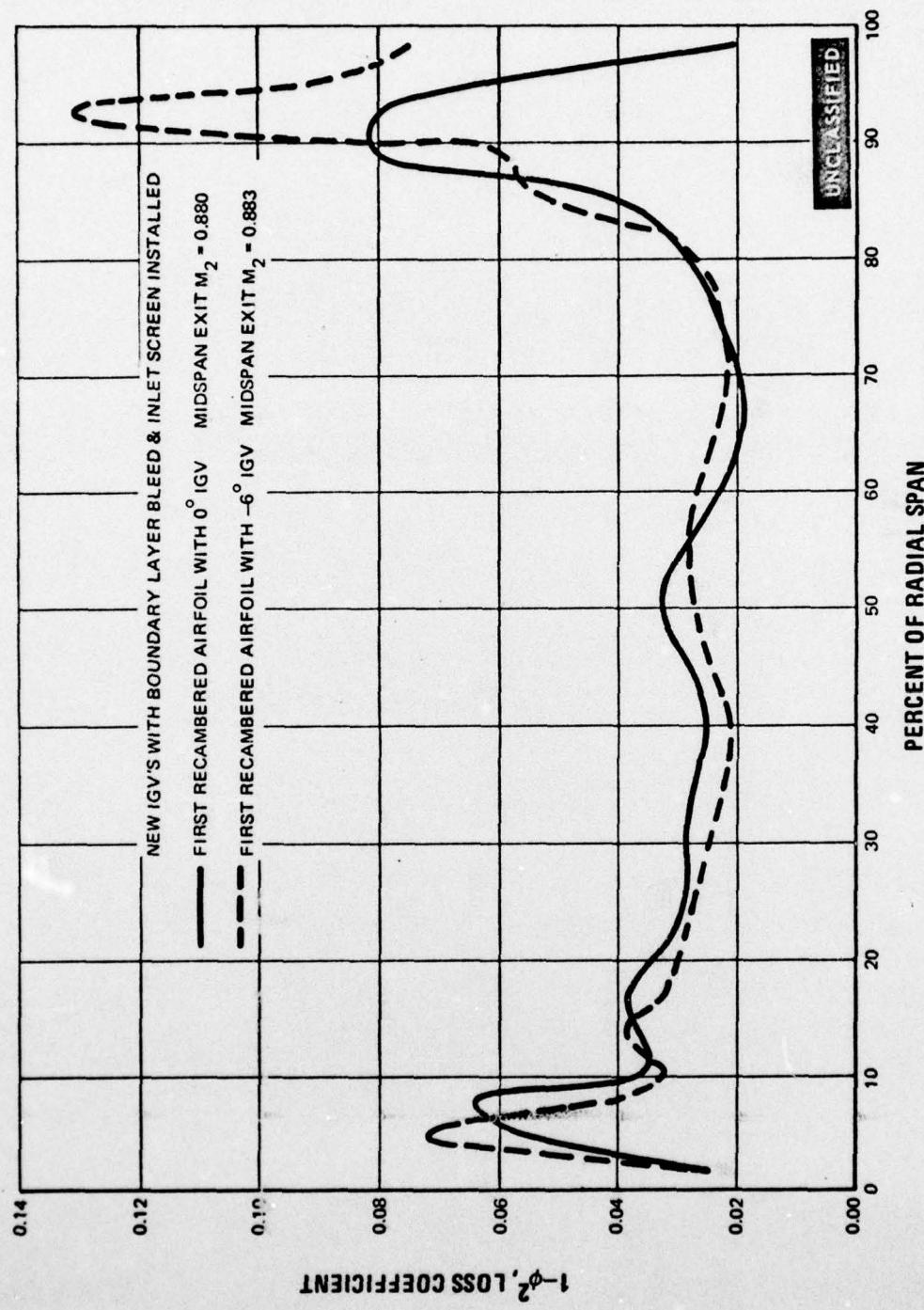


Figure 171 Comparison of First Recambered Airfoil Loss Coefficient Distribution at  
-6 Degrees Incidence With Zero Incidence Values

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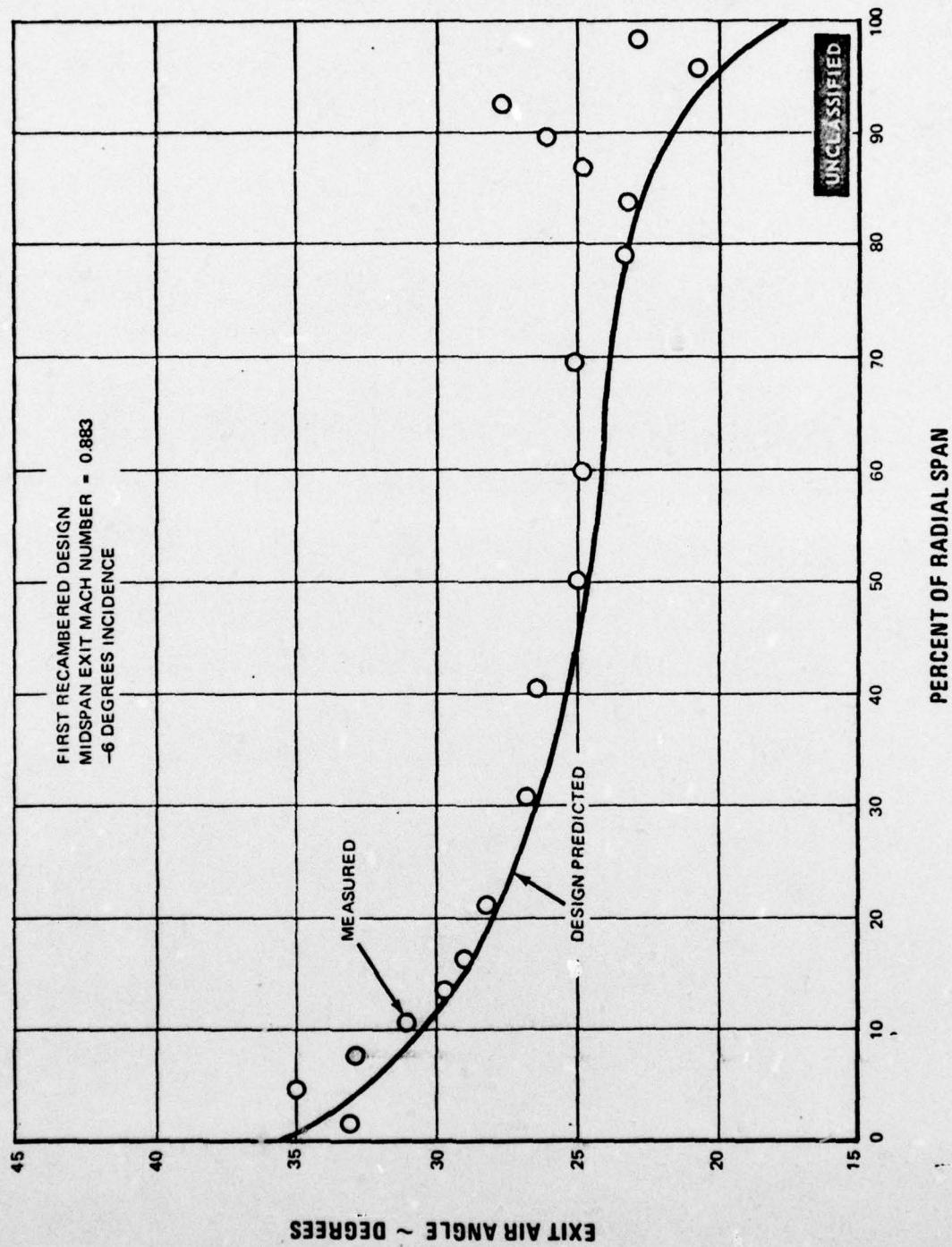


Figure 172 Comparison of Measured Exit Gas Angle With Predicted Design Values at -6 Degrees Incidence. First Recambered Airfoil at Midspan Exit Mach No. = 0.883

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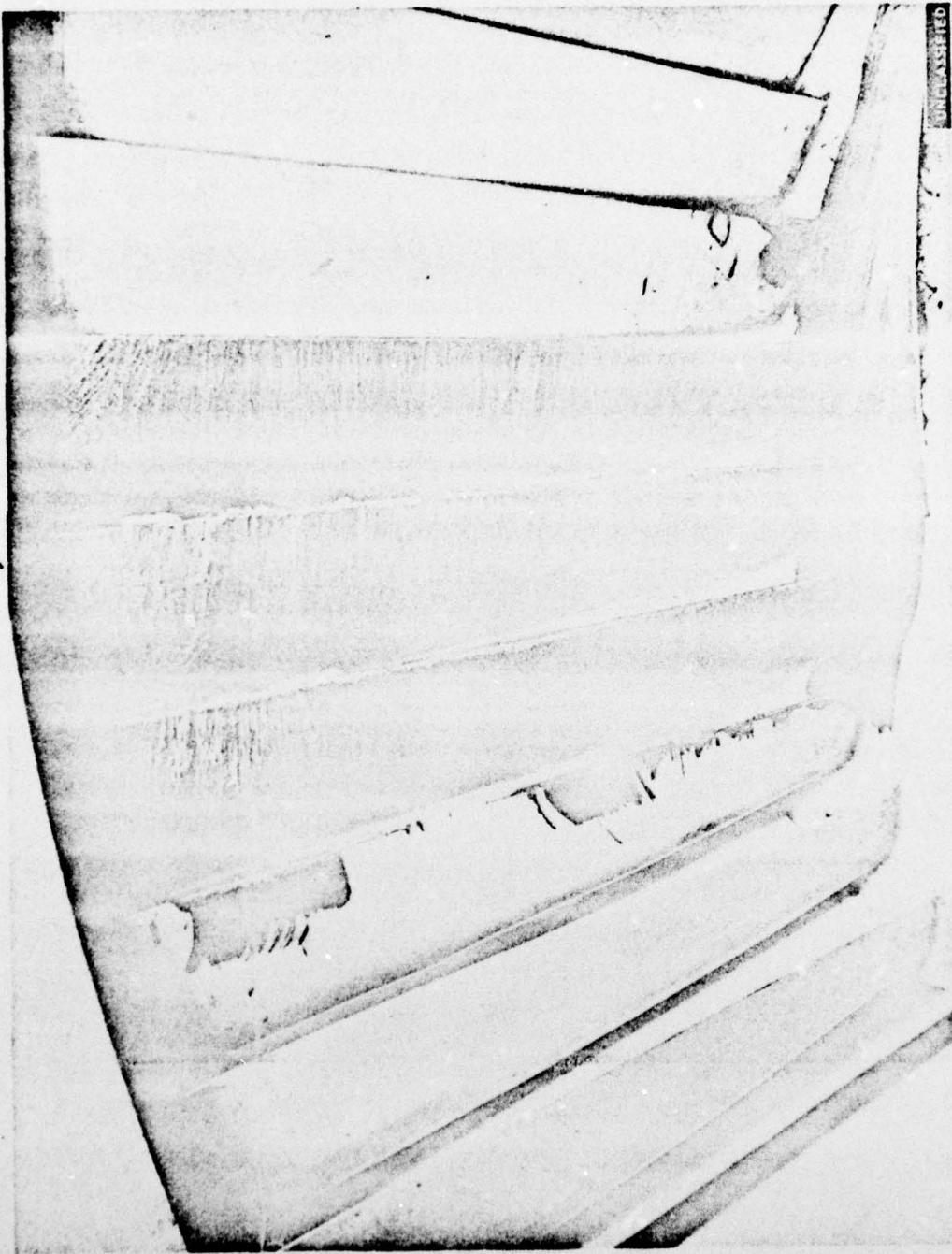


Figure 173 Oil and Graphite Flow Patterns, First Recambered Airfoil at +10 Degrees Incidence. Midspan Exit Mach No. = 0.840 (FE95376)

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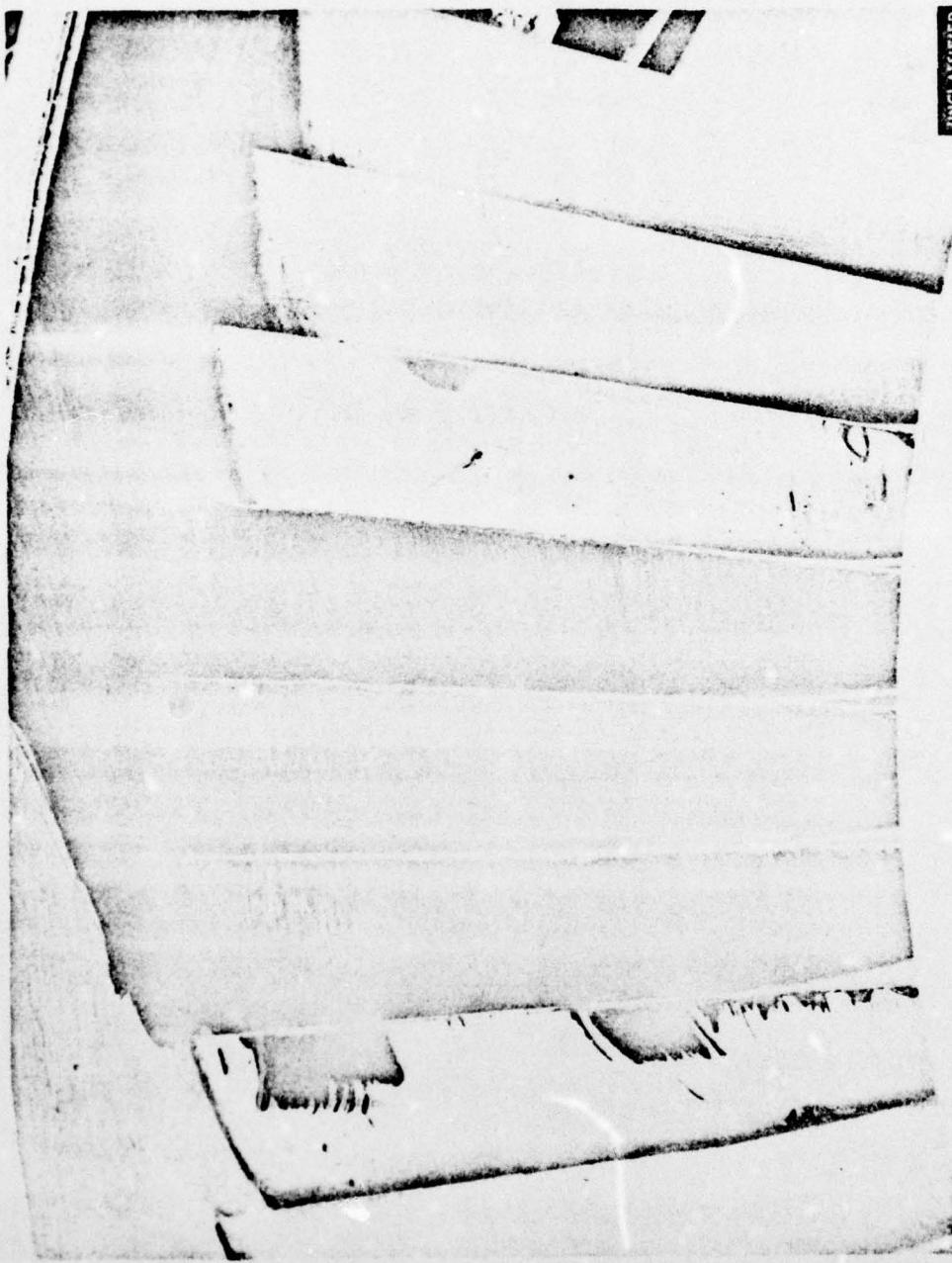
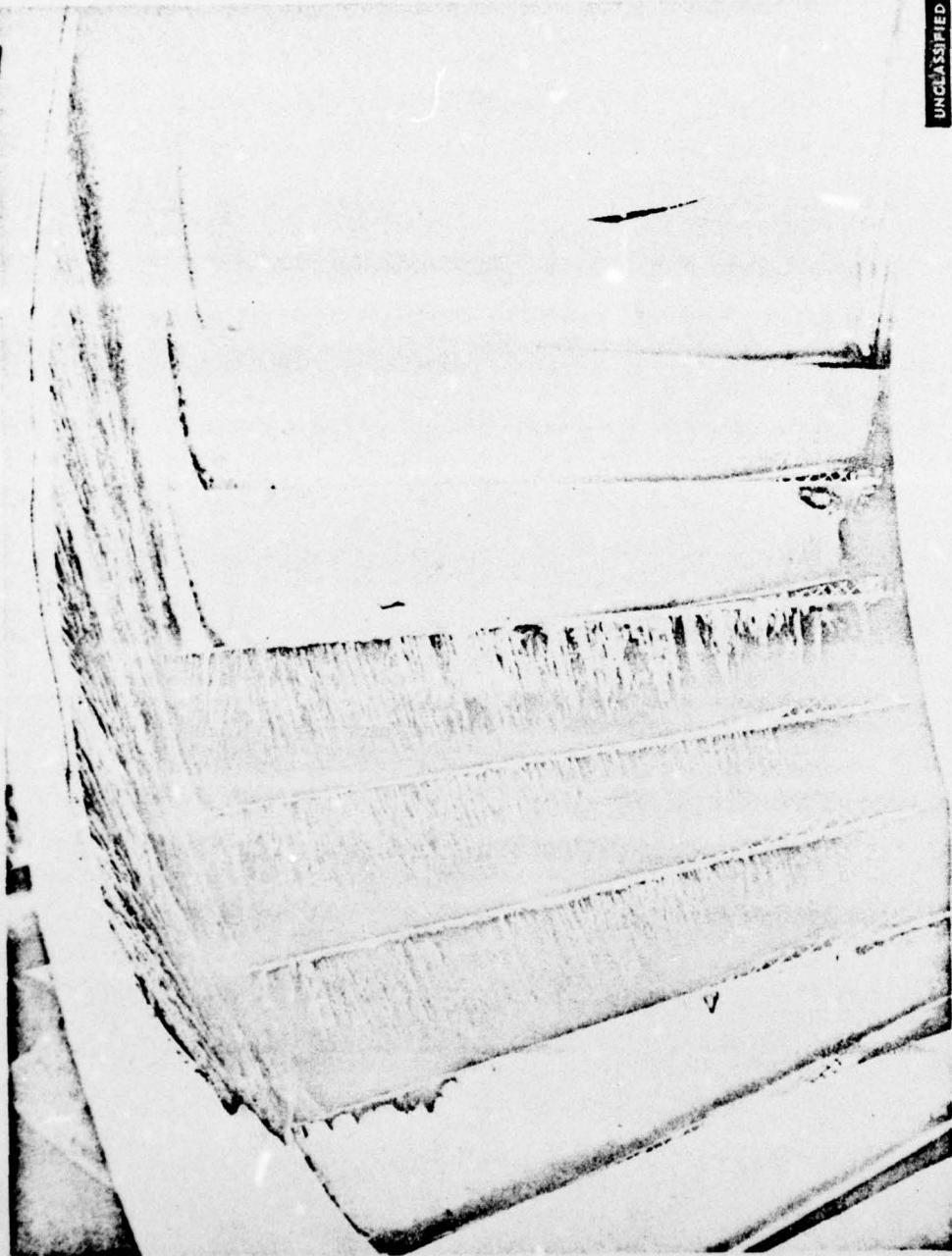


Figure 174 Oil and Graphite Flow Patterns, First Recambered Airfoil at +10 Degrees Incidence. Midspan Exit Mach No. = 0.840 (FE95377)

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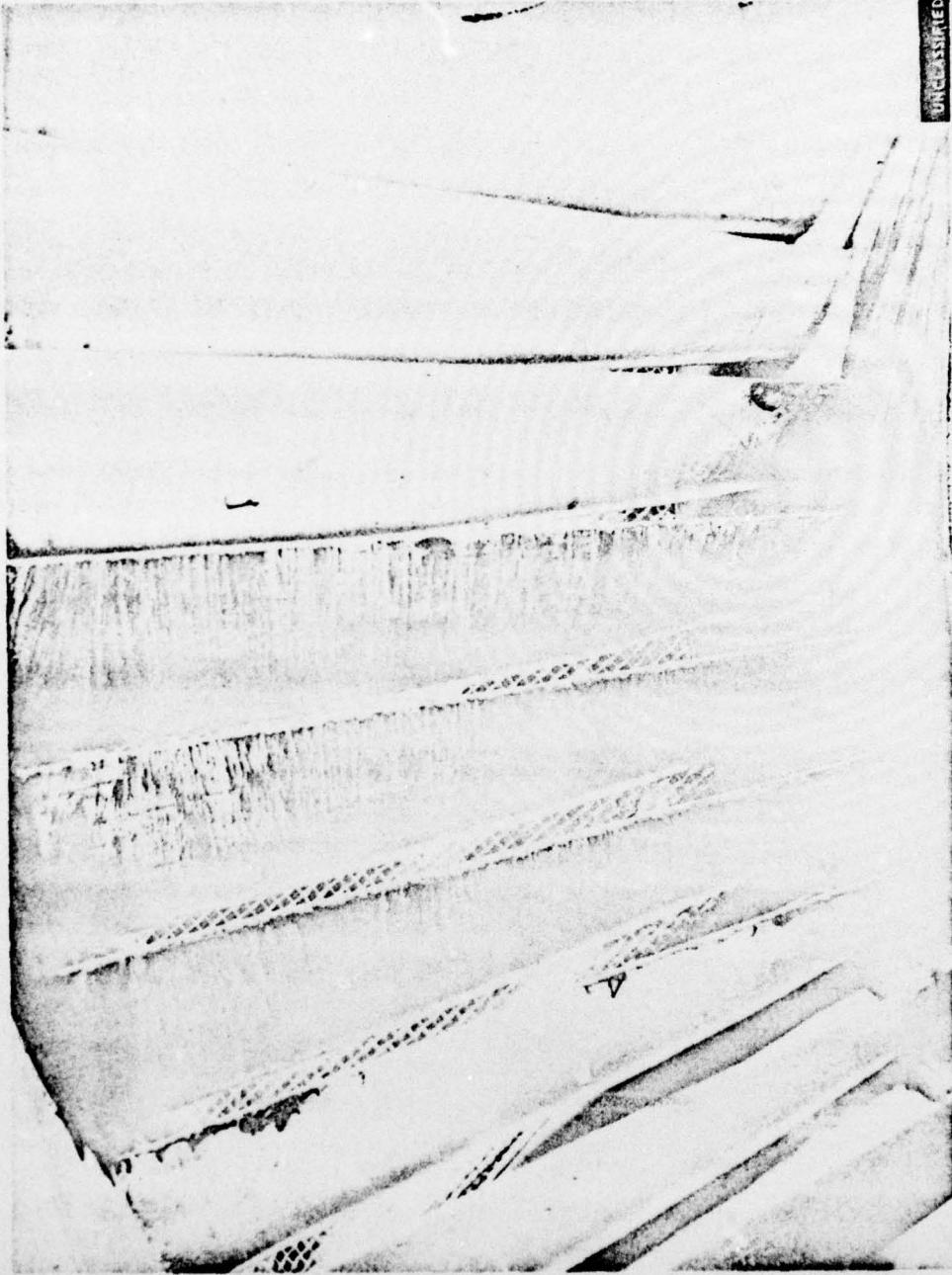


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Figure 175 Oil and Graphite Flow Patterns, First Recambered Airfoil at -6 Degrees Incidence. Midspan Exit Mach No. = 0.883 (FE95504)

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Figure 176 Oil and Graphite Flow Patterns, First Recambered Airfoil at -6 Degrees Incidence. Midspan Exit Mach No. = 0.883 (FE95503)

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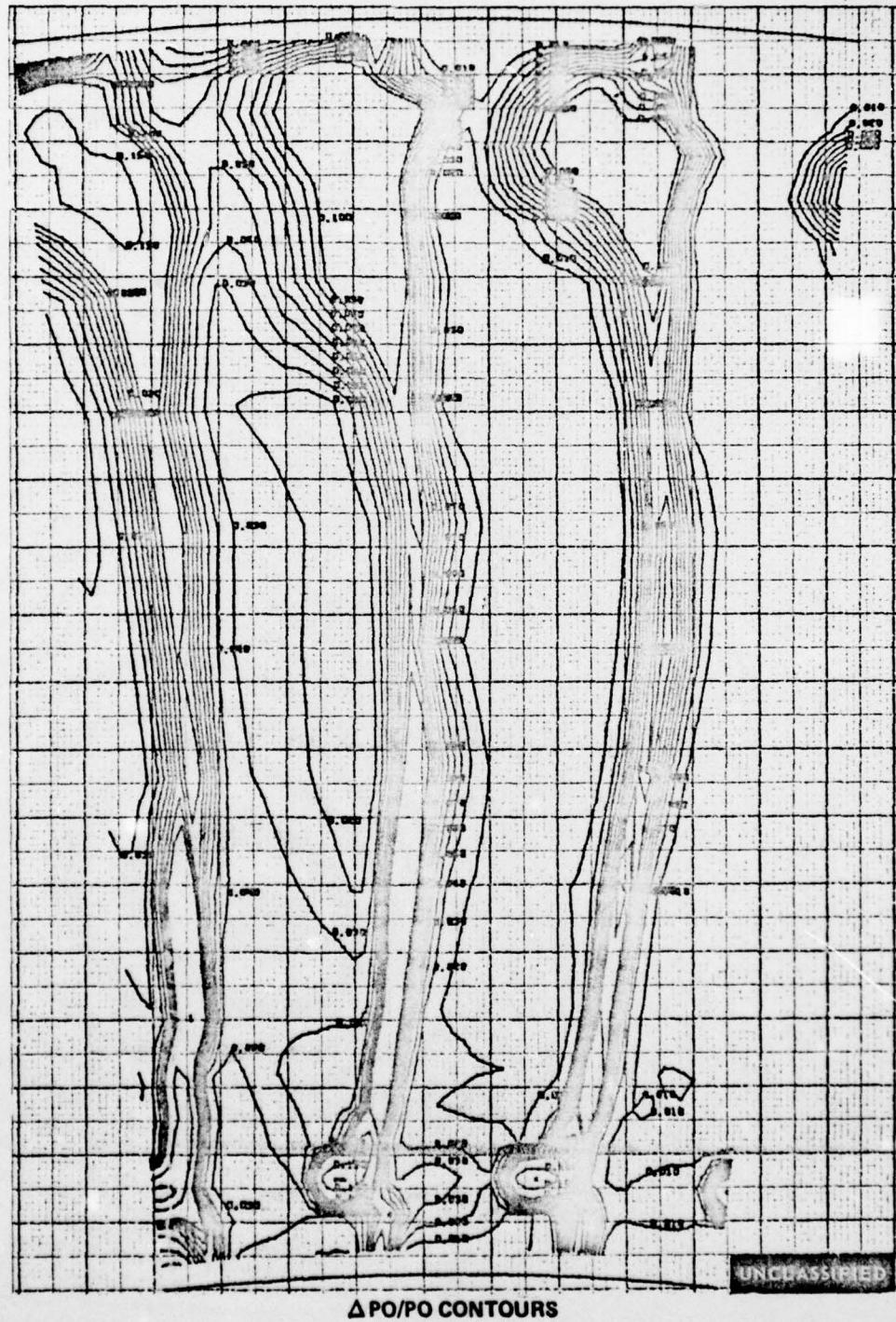
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(U) Performance data for the second vane recambering design A airfoil are shown in Figures 177 through 182 at the positive incidence angle of 10 degrees. Similar data for a negative incidence of 6 degrees are shown in Figures 183 through 188. Spanwise loss coefficient distributions at +10 degrees incidence and -6 degrees incidence are compared to zero degrees incidence values in Figures 189 and 190, respectively. The root end-wall losses were larger for positive incidence and showed little difference for the negative incidence, when compared to the zero incidence results. The tip end-wall losses were much larger at the positive incidence and slightly lower at the negative incidence, when compared to the zero incidence condition. Overall, the positive incidence loss was greater and the negative incidence less when compared to the zero incidence curve. The measured exit gas angles are shown in Figures 191 and 192 for the +10 degrees and -6 degrees evaluations, respectively. Slight underturning is indicated at the tip in both cases with good agreement with design predicted values over the rest of the airfoils. This was also true at the zero incidence test condition (Section V). Oil and graphite flow patterns are shown in Figures 193 through 196 for both positive and negative incidence tests.

(U) The last airfoil subjected to off-design evaluation was the second vane recambering design C. The performance data for this airfoil at +10 degrees incidence are presented in Figures 197 through 202. The same type of data for the -6 degree incidence is shown in Figures 203 through 208. Comparisons of the spanwise loss coefficient distributions at positive incidence and negative incidence are made with zero incidence values in Figures 209 and 210, respectively. The root end-wall losses were slightly lower at the +10 degrees and -6 degrees incidence than at the zero incidence condition. The positive incidence tip end-wall loss was greater than, and the negative incidence tip end-wall loss was about the same as the zero incidence value. The overall loss coefficient was slightly higher at the positive incidence, and slightly lower at the negative incidence, in comparison to zero incidence conditions. The measured exit gas angles are shown in Figures 211 and 212. Good agreement was obtained with design values for both positive and negative incidence angles. Oil and graphite flow patterns for these tests are shown in Figures 213 through 216. Well behaved flow with some end-wall radial shift, is again verified from these patterns.

(U) The variation of loss coefficient with midspan exit Mach number is shown in Figures 217, 218 and 219 for the second vane first recambered airfoil, recambering design A and recambering design C, respectively. The contribution of each end to the overall loss is shown, in order to indicate the effect of incidence on each end-wall uncambering. For these same airfoils, the effect of incidence on loss coefficient is shown in Figure 220. It is noted that these airfoils were not overly sensitive to incidence, and that the loss level generally goes up with incidence angle.

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**Figure 177** Pressure Loss Contours, Second Vane Recambering Design A at +10° Incidence. Three Flow Passages. Midspan Exit Mach Number = 0.837

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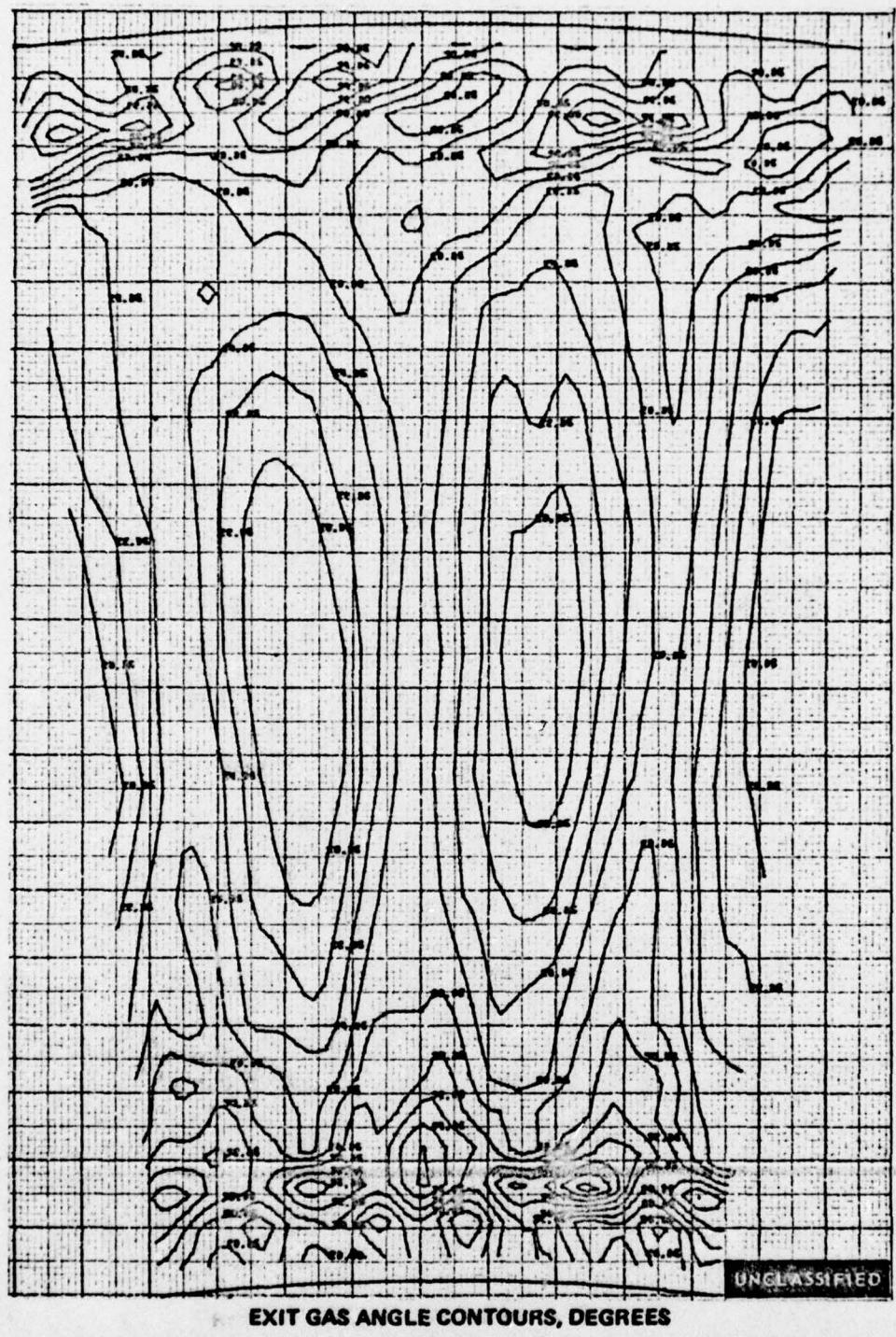
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INVESTIGATION OF A HIGHLY LOADED TWO-STAGE FAN-DRIVE TURBINE. V--ETC(U)  
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**Figure 178** Exit Gas Angle Contours, Second Vane Recambering Design A at +10° Incidence. Three Flow Passages. Midspan Exit Mach Number = 0.837

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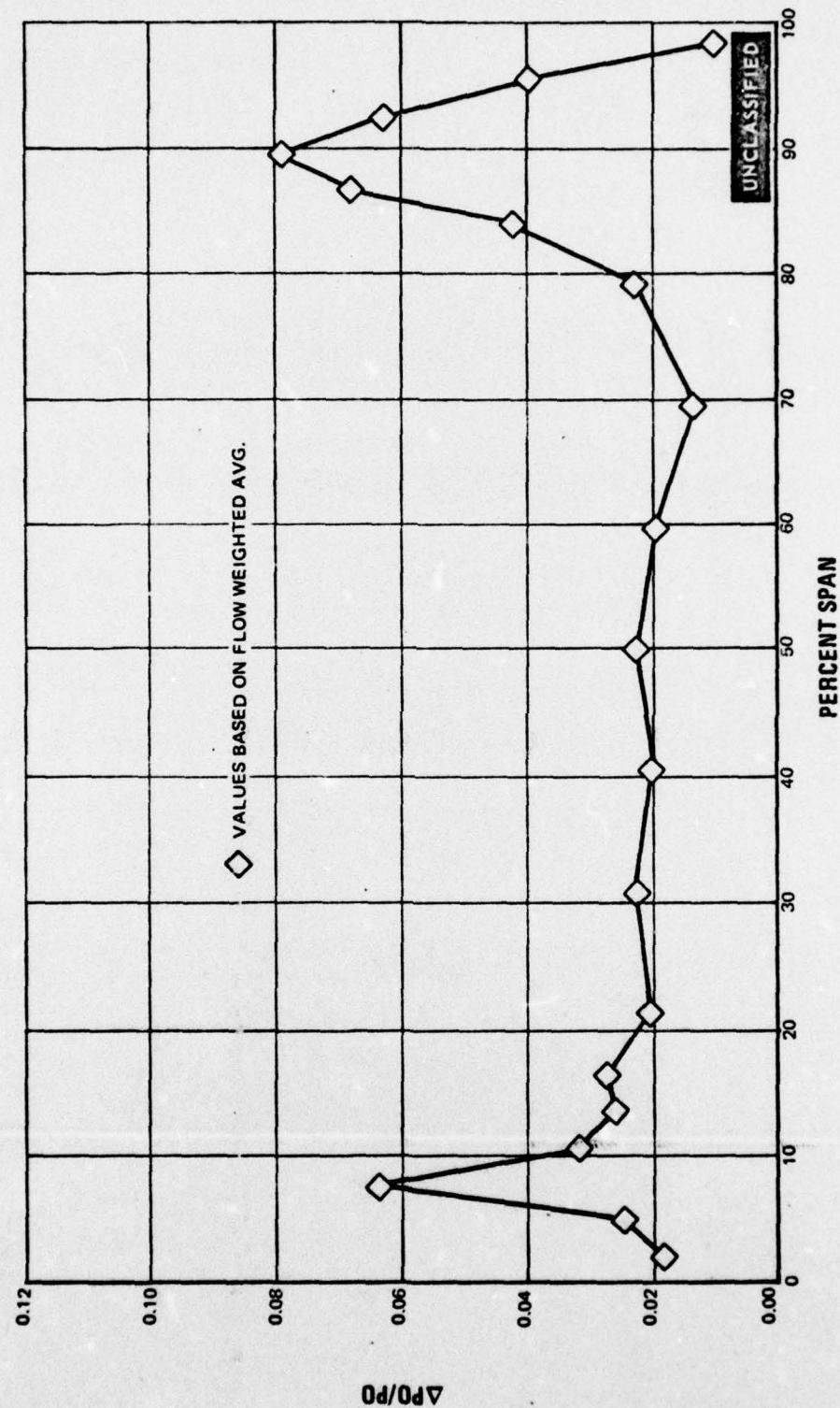


Figure 179 Spanwise Pressure Loss Distribution, Second Vane Recambering Design A  
at  $+10^\circ$  Incidence. Midspan Exit Mach Number = 0.837

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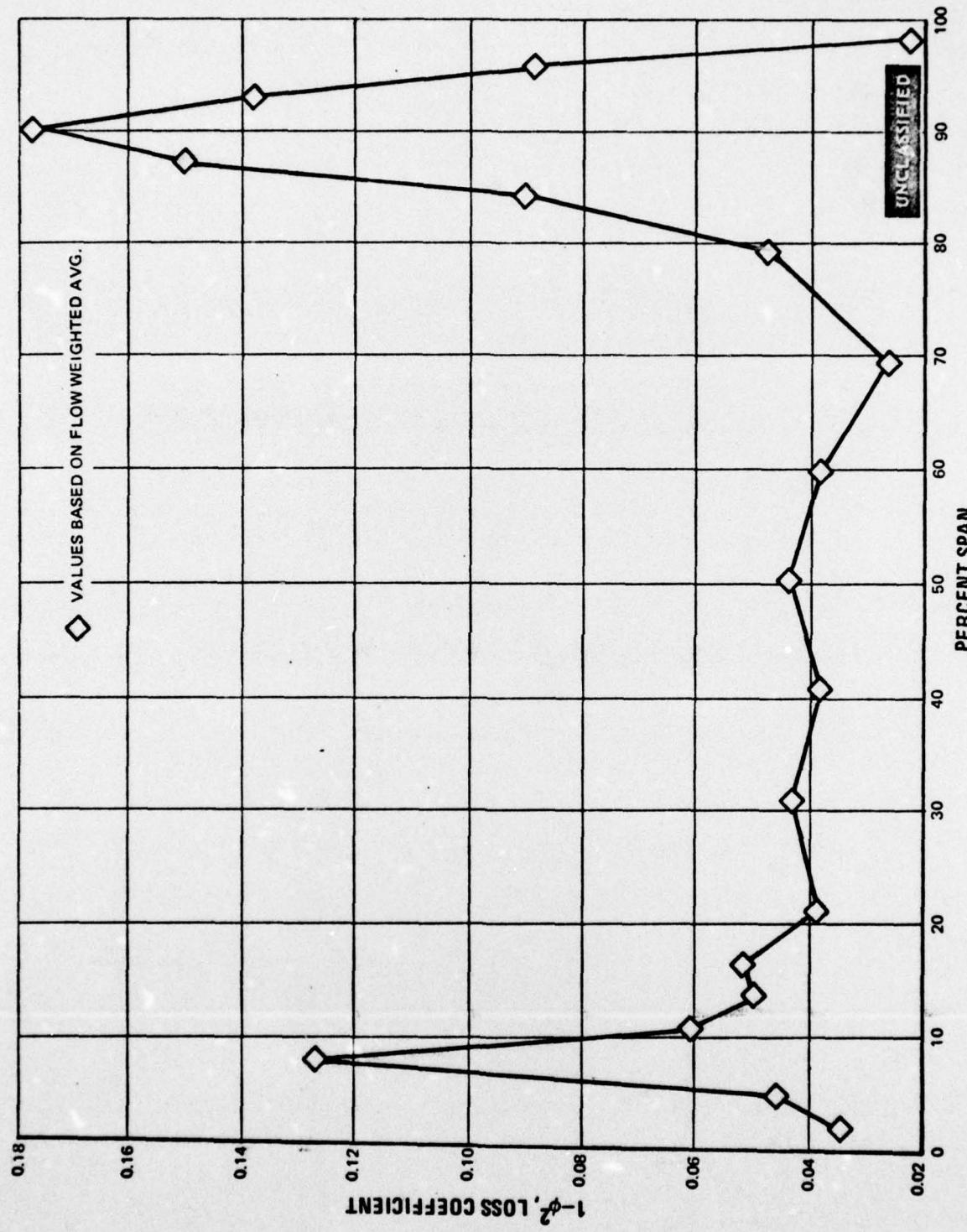


Figure 180 Spanwise Loss Coefficient Distribution, Second Vane Recambering Design A  
at  $15^\circ$  Incidence. Midspan Exit Mach Number = 0.837

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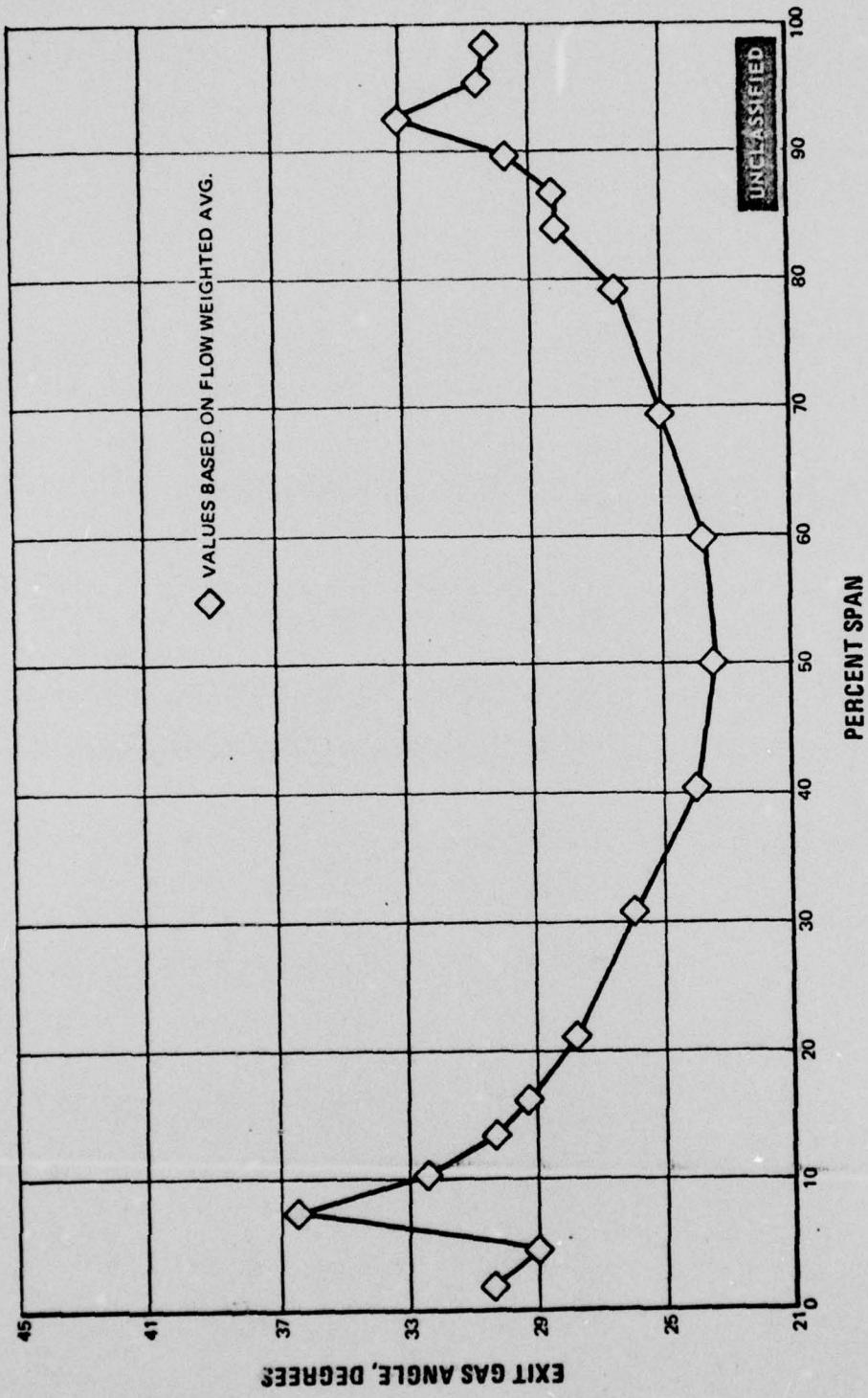


Figure 181 Spanwise Exit Gas Angle Distribution, Second Vane Recambering Design A  
at +10° Incidence. Midspan Exit Mach Number = 0.837

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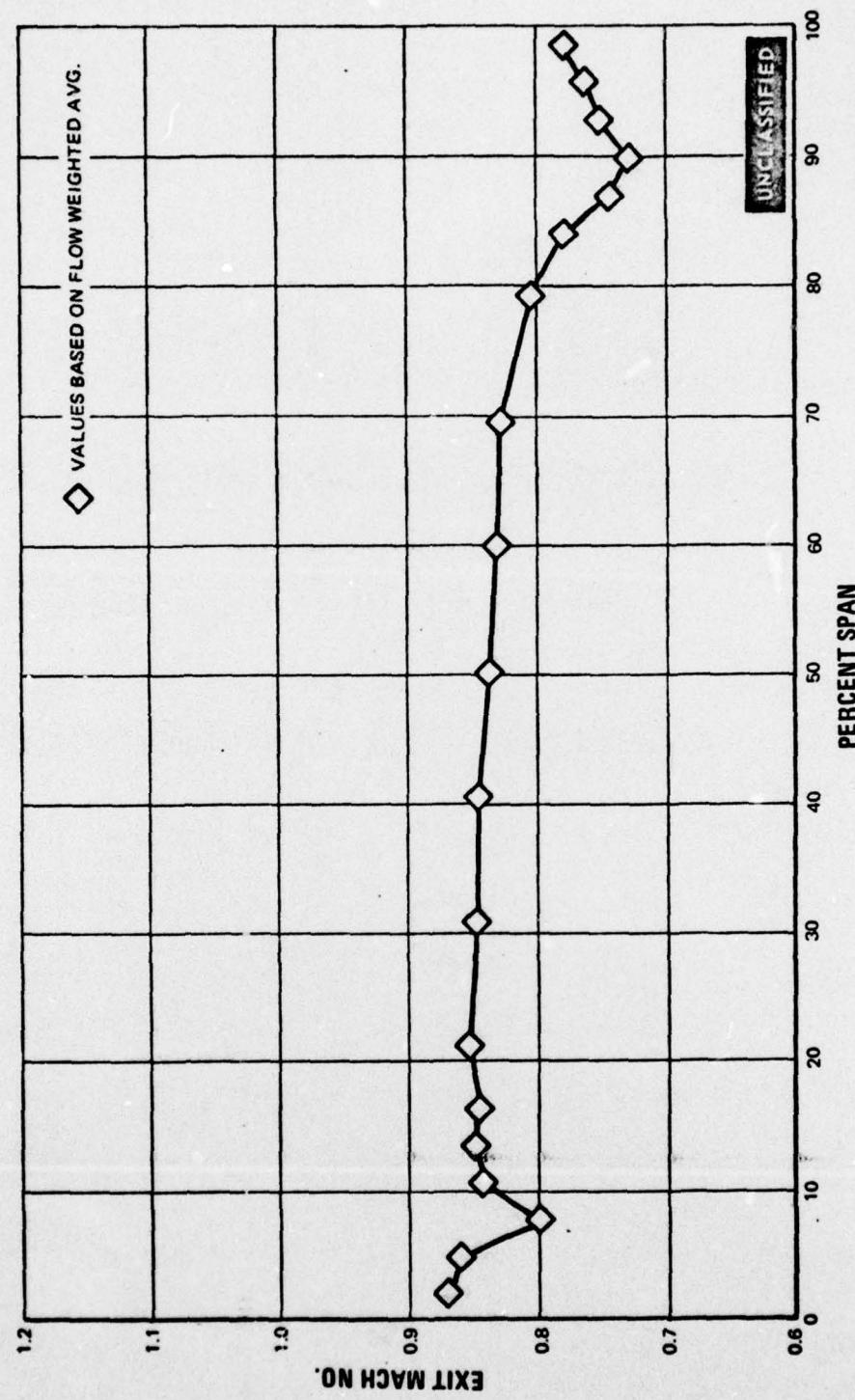
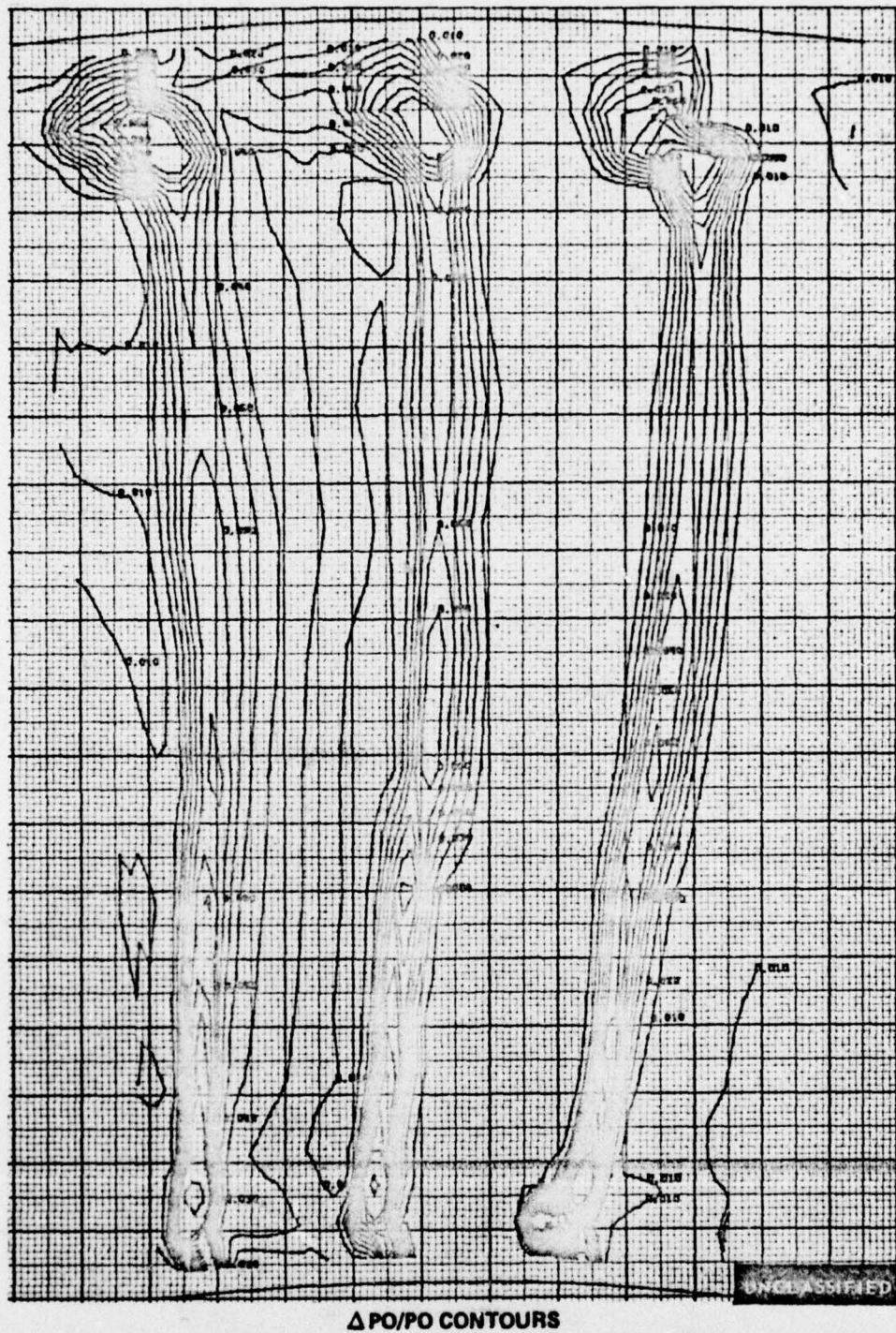


Figure 182 Spanwise Exit Mach Number Distribution, Second Vane Recambering Design  
A at +10° Incidence, Midspan Exit Mach Number = 0.837

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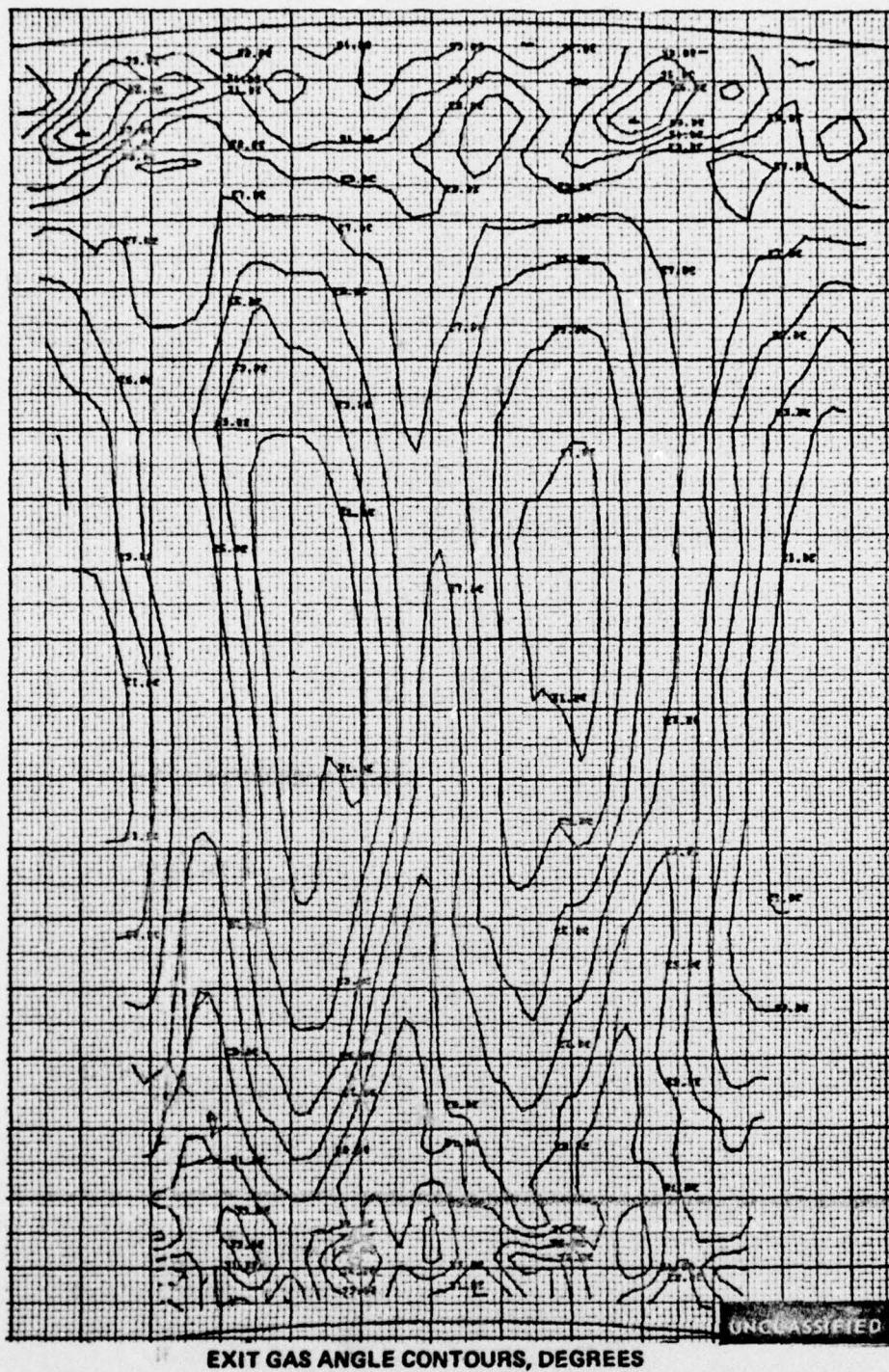
**UNCLASSIFIED**



**Figure 183 Pressure Loss Contours, Second Vane Recambering Design A at  $-6^\circ$  Incidence.  
Three Flow Passages. Midspan Exit Mach Number = 0.869**

**UNCLASSIFIED**

**UNCLASSIFIED**



**Figure 184** Exit Gas Angle Contours, Second Vane Recambering Design A at  $-6^\circ$  Incidence. Three Flow Passages. Midspan Exit Mach Number = 0.869

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**UNCLASSIFIED**

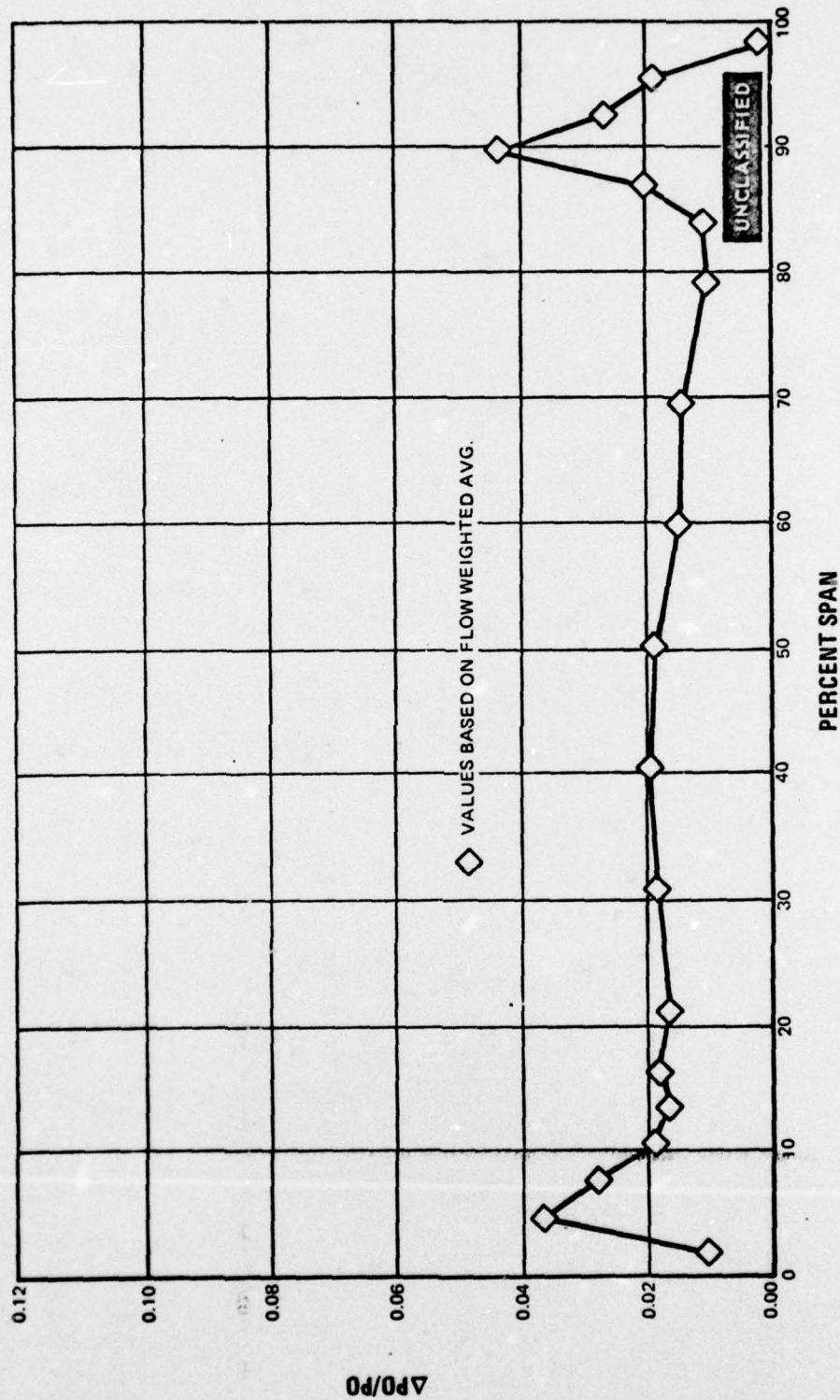


Figure 185 Spanwise Pressure Loss Distribution, Second Vane Recambering Design A at  
-6° Incidence. Midspan Exit Mach Number = 0.869

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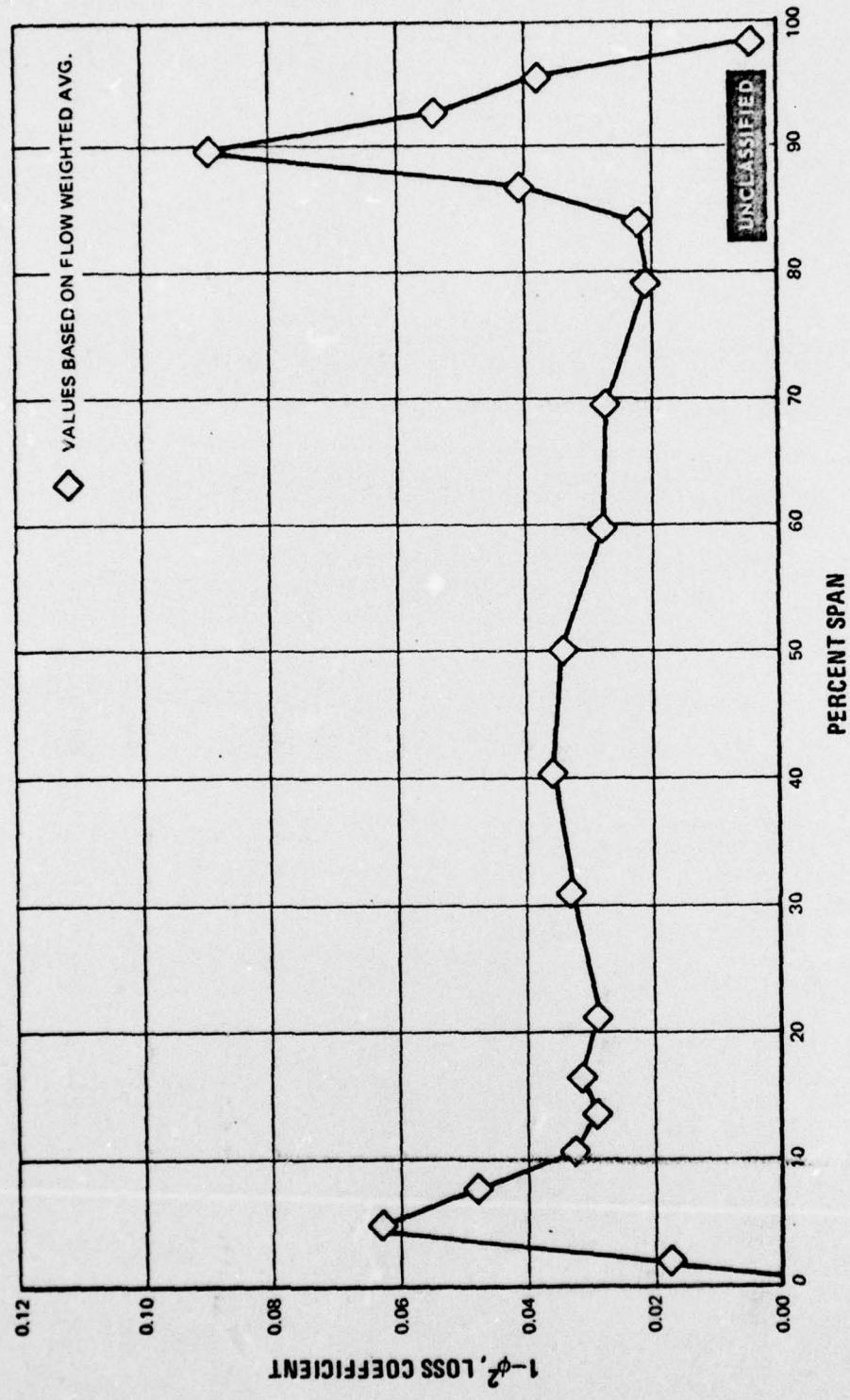


Figure 186 Spanwise Loss Coefficient Distribution, Second Vane Recambering Design A  
at -6° Incidence. Midspan Exit Mach Number = 0.869

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**UNCLASSIFIED**

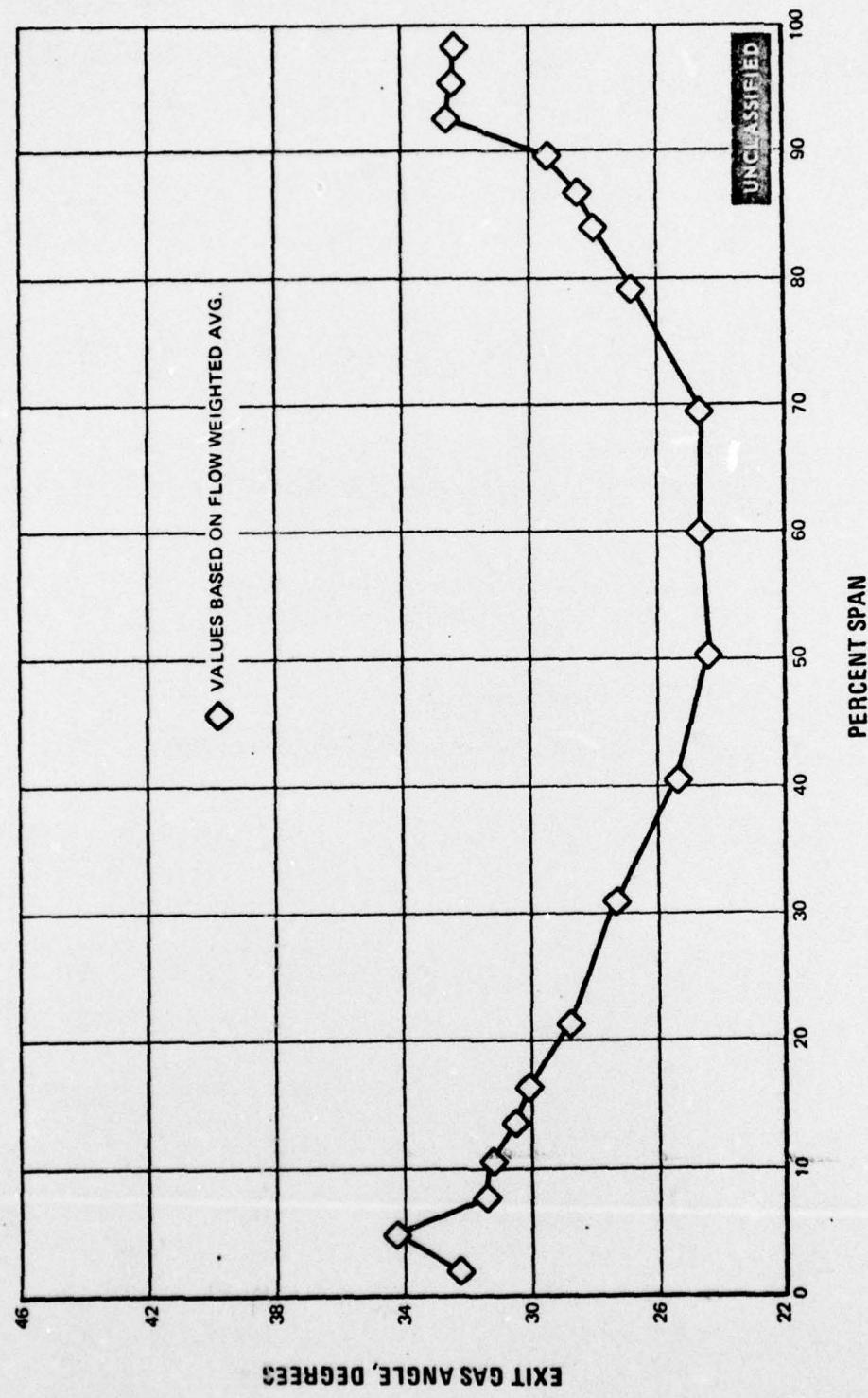


Figure 187 Spanwise Exit Gas Angle Distribution, Second Vane Recambering Design A  
at -6° Incidence. Midspan Exit Mach Number = 0.869

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**UNCLASSIFIED**

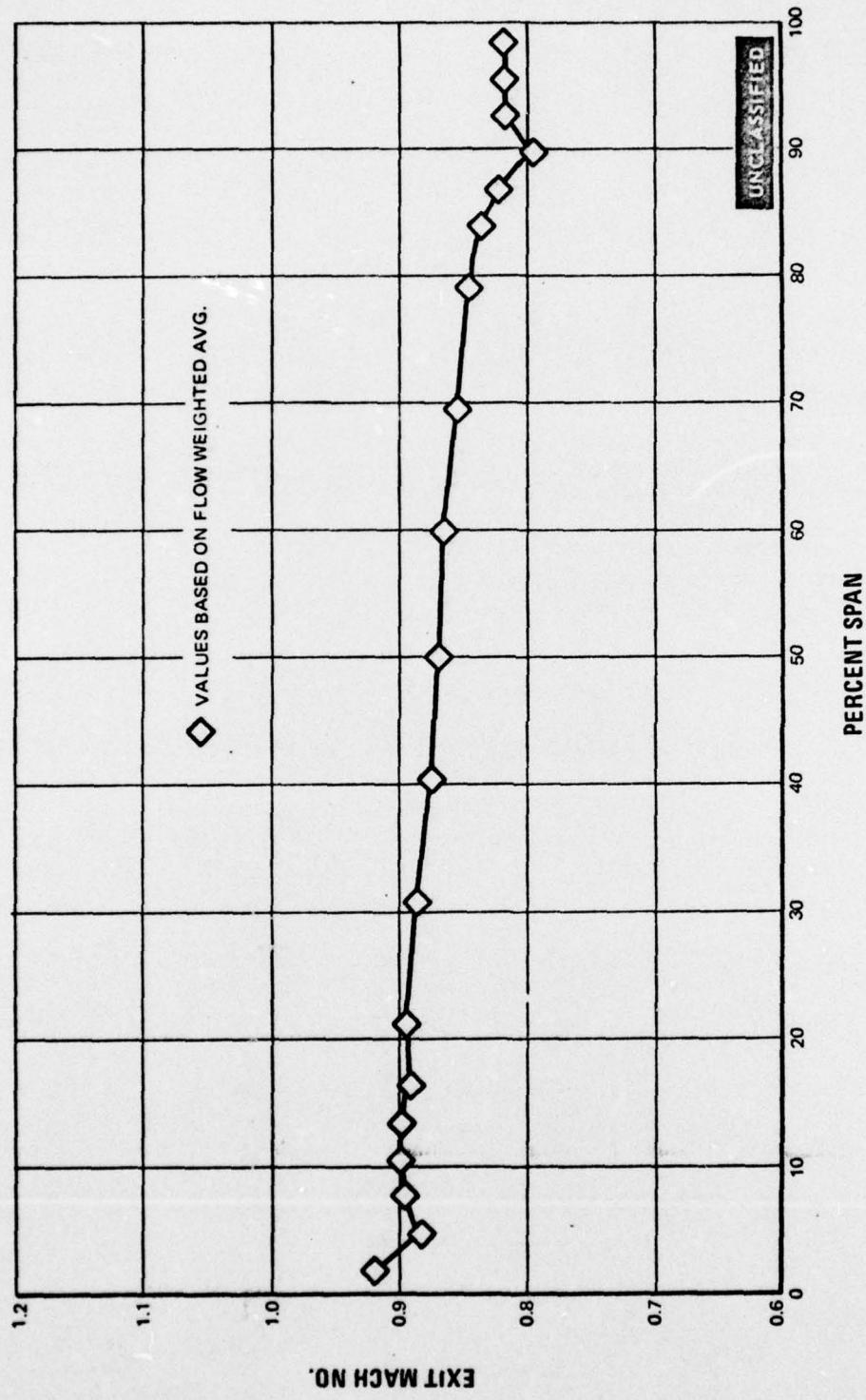


Figure 188 Spanwise Exit Mach Number Distribution, Second Vane Recambering Design  
A at -6° Incidence. Midspan Exit Mach Number = 0.869

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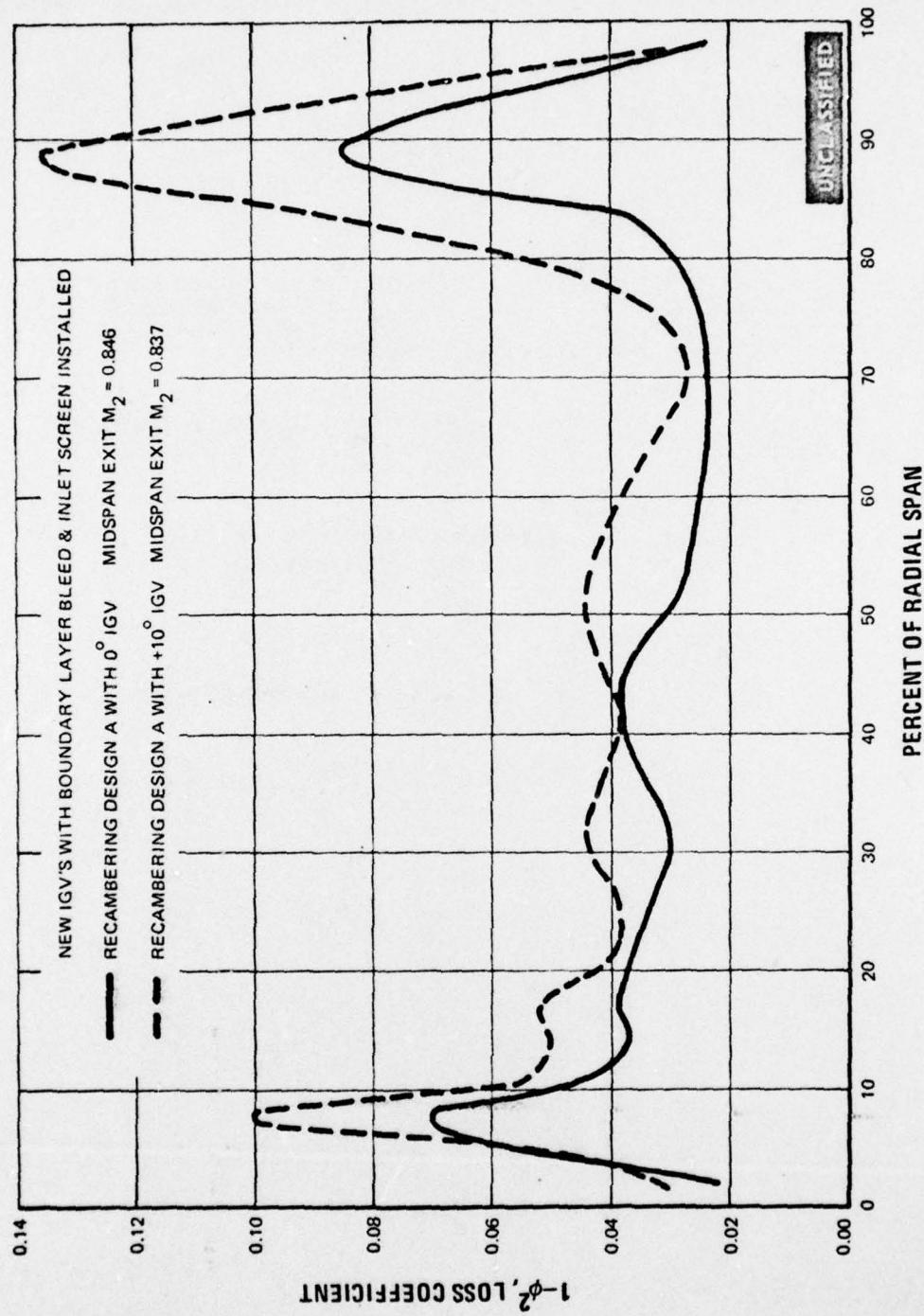


Figure 189 Comparison of Recambering Design A Airfoil Loss Coefficient Distribution at +10 Degrees Incidence With Zero Incidence Values

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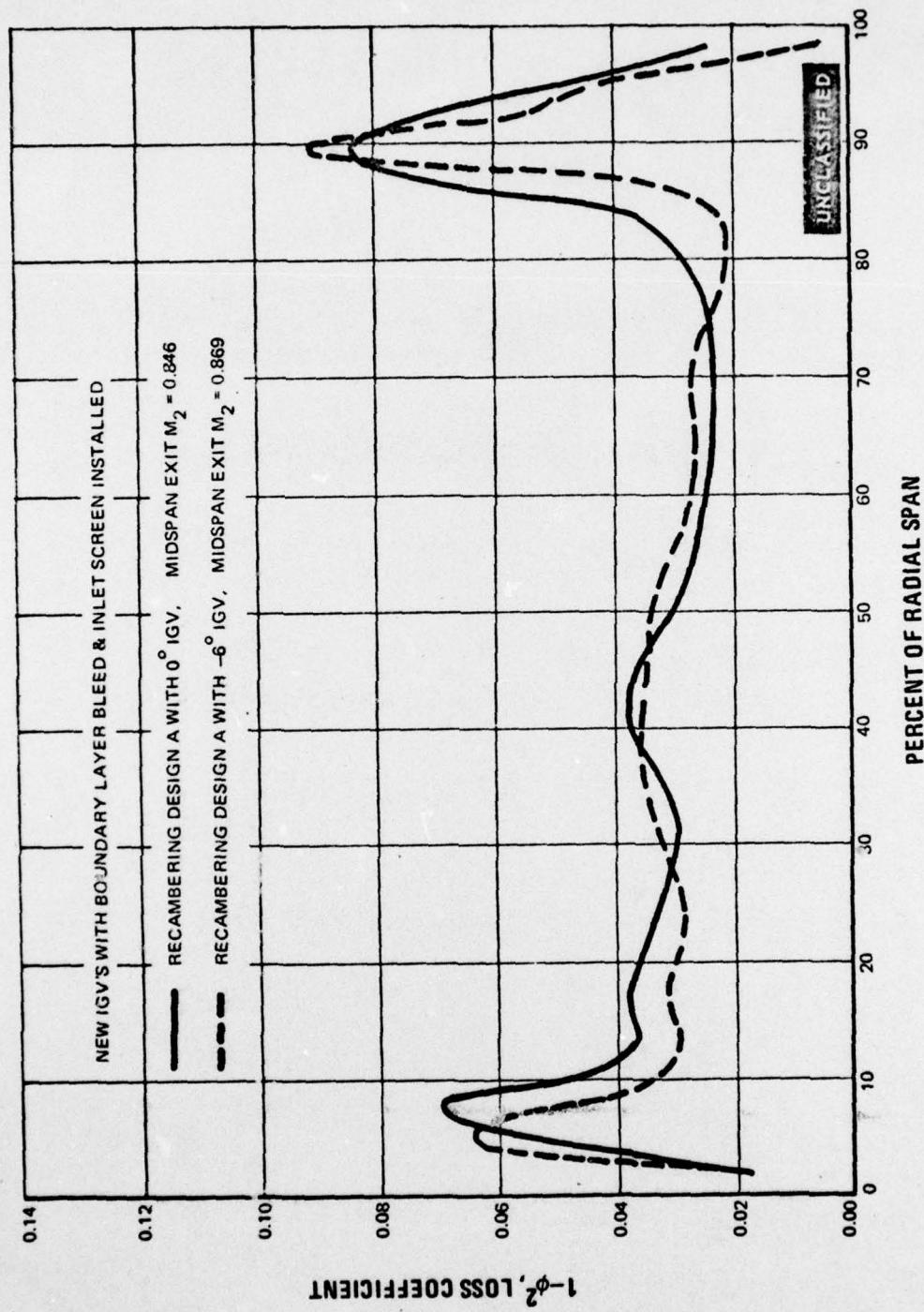


Figure 190 Comparison of Recambering Design A Airfoil Loss Coefficient Distribution  
at -6 Degrees Incidence With Zero Incidence Values

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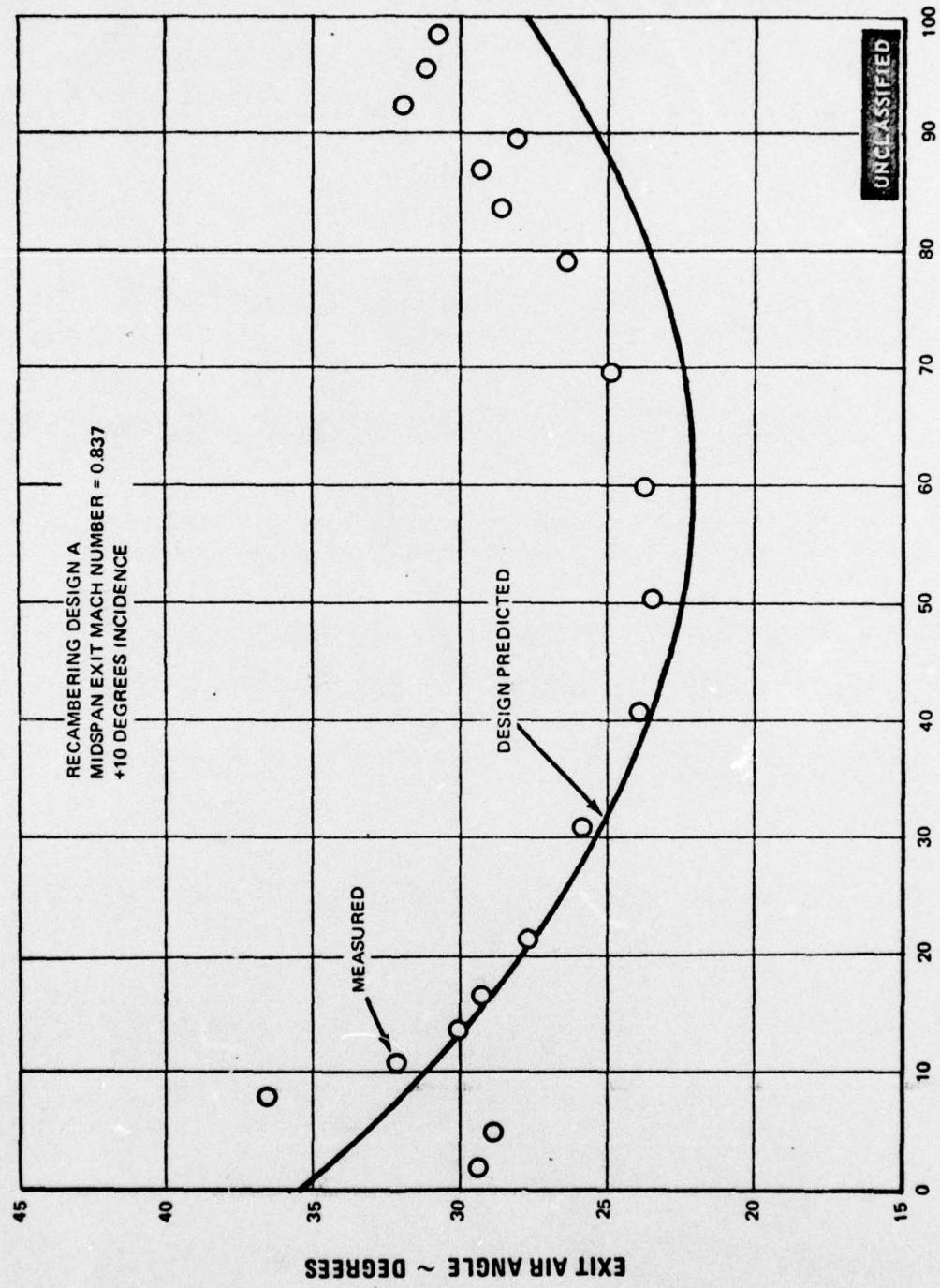


Figure 191 Comparison of Measured Exit Gas Angle With Predicted Design Values at +10 Degrees Incidence. Design A Airfoil at Midspan Exit Mach No. = 0.837

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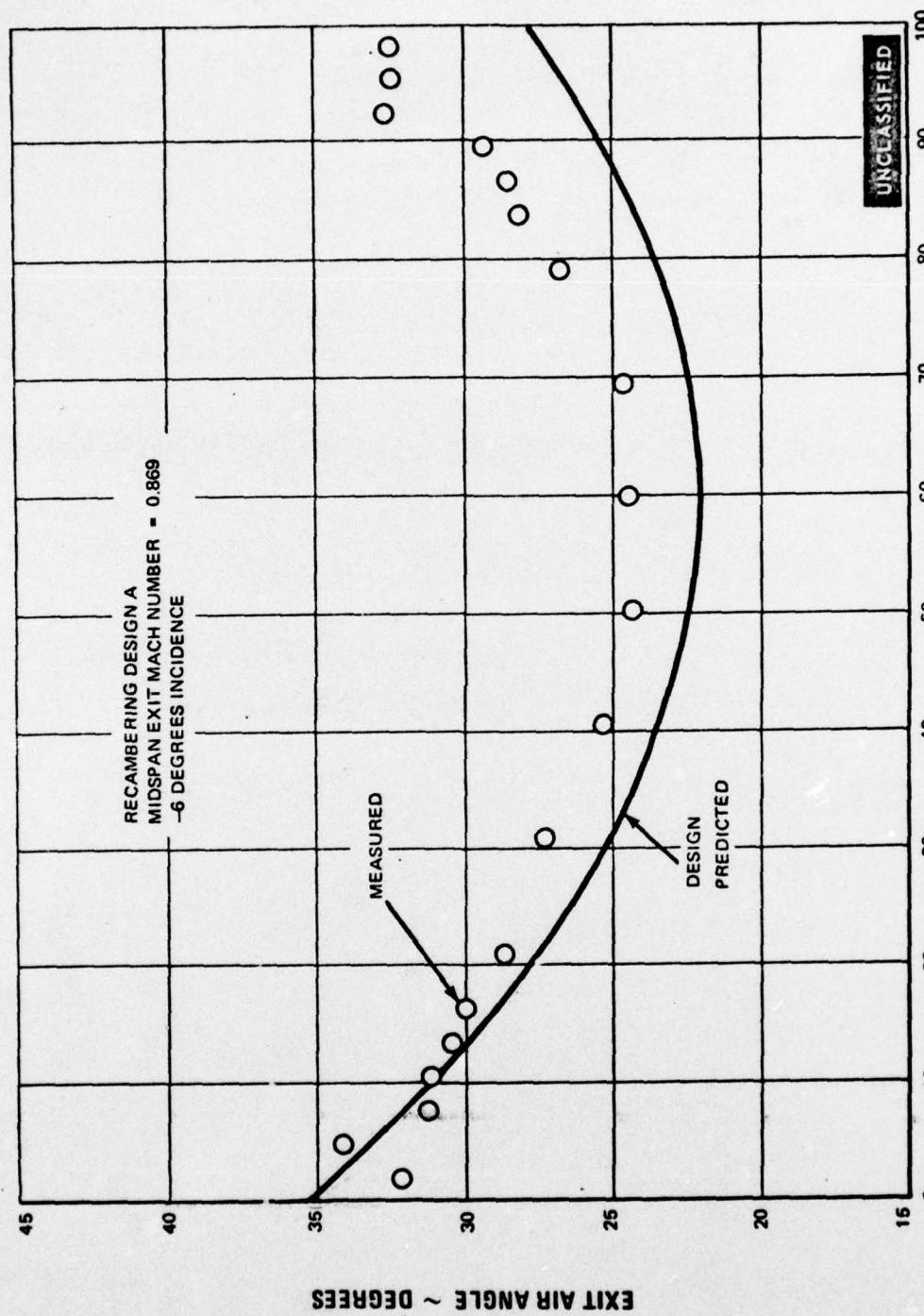


Figure 192 Comparison of Measured Exit Gas Angle With Predicted Design Values at -6 Degrees Incidence. Design A Airfoil at Midspan Exit Mach No. = 0.869

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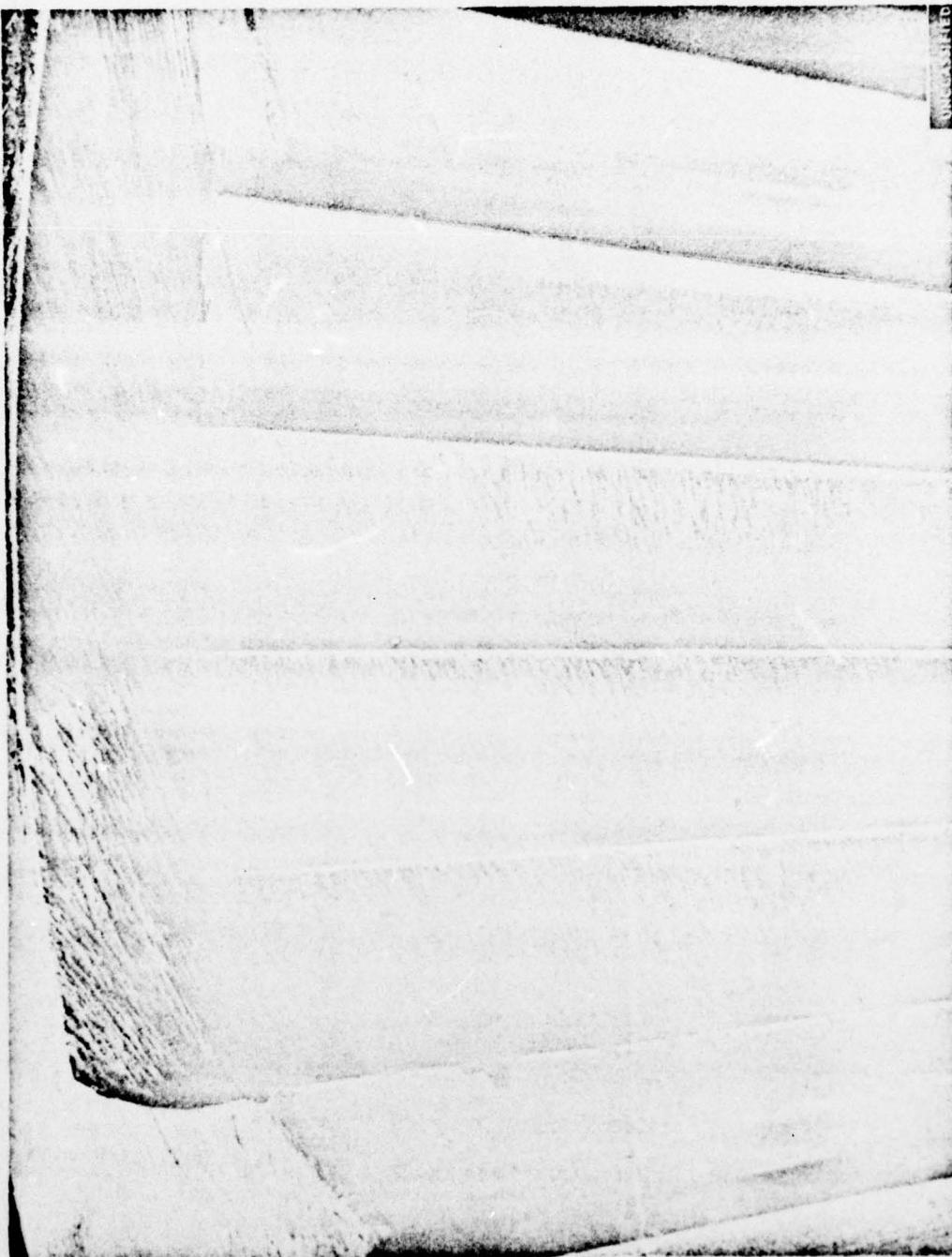


Figure 193 O! and Crzwhite Flow Patterns, Second Vane Recambering Design A Airfoil  
at +10 Degrees Incidence. Midspan Exit Mach No. = 0.837 (FE95732)

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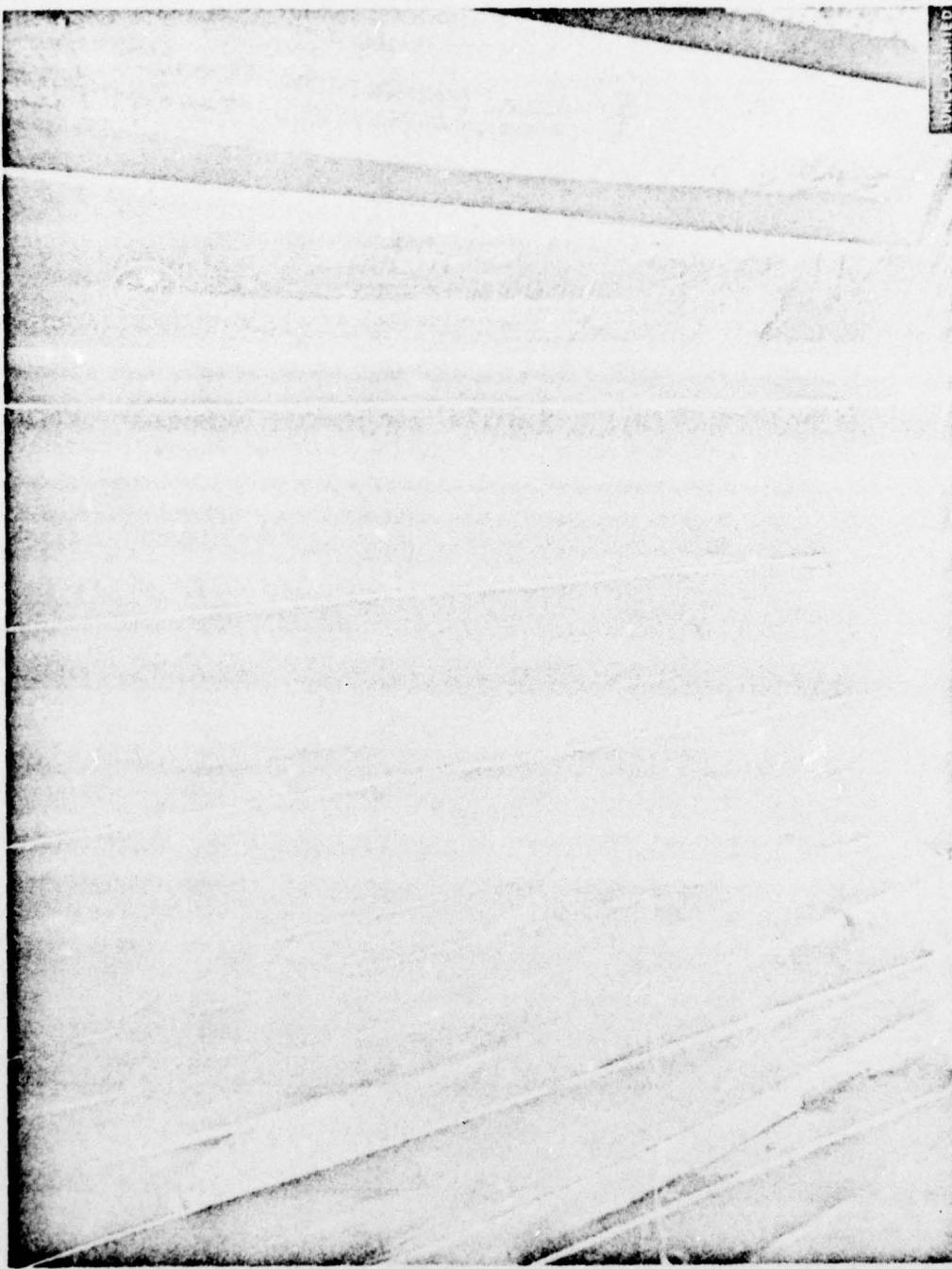
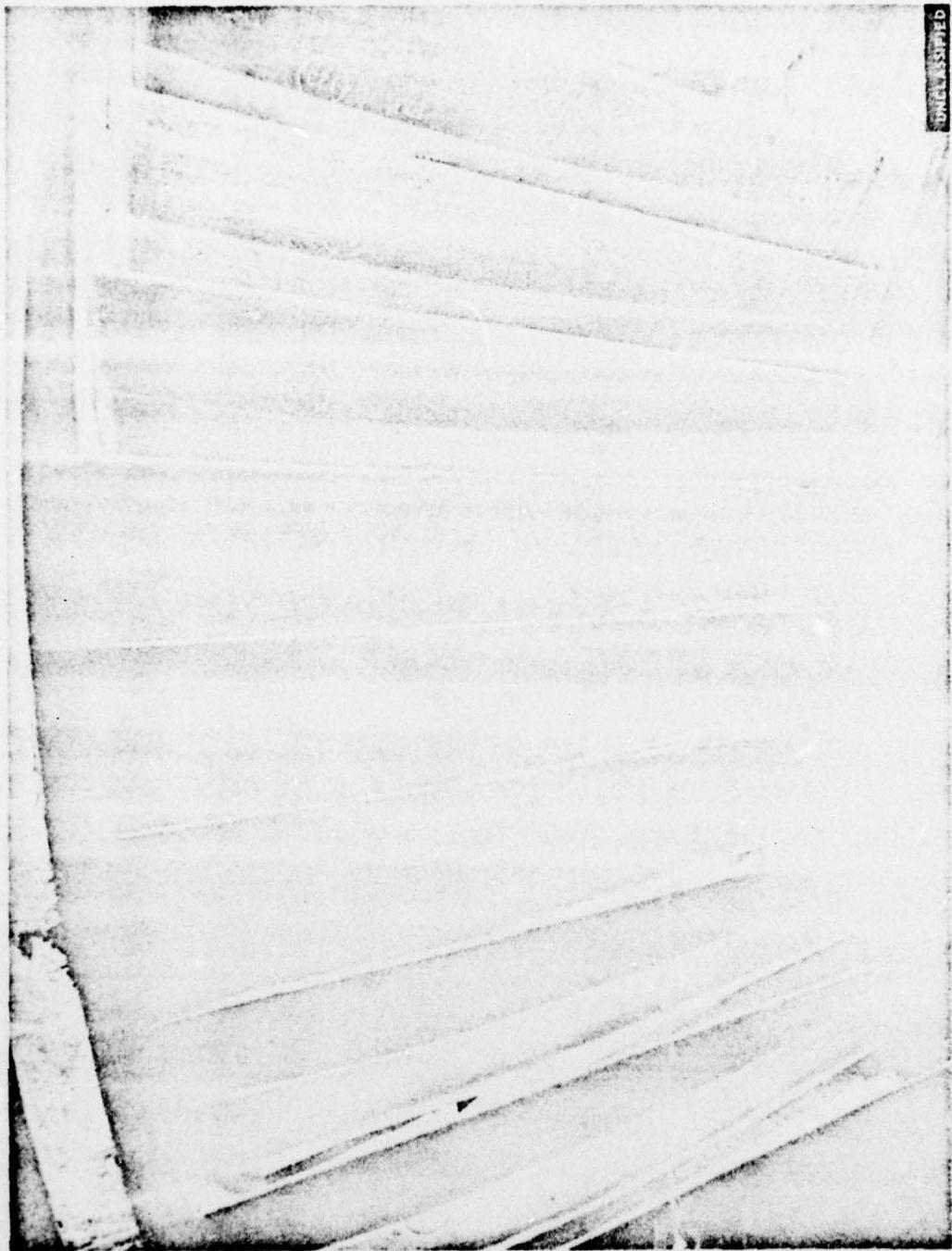


Figure 194 Oil and Graphite Flow Patterns, Second Vane Recambering Design A Airfoil  
at  $\pm 10$  Degrees Incidence. Midspan Exit Mach No. = 0.837 (FE95733)

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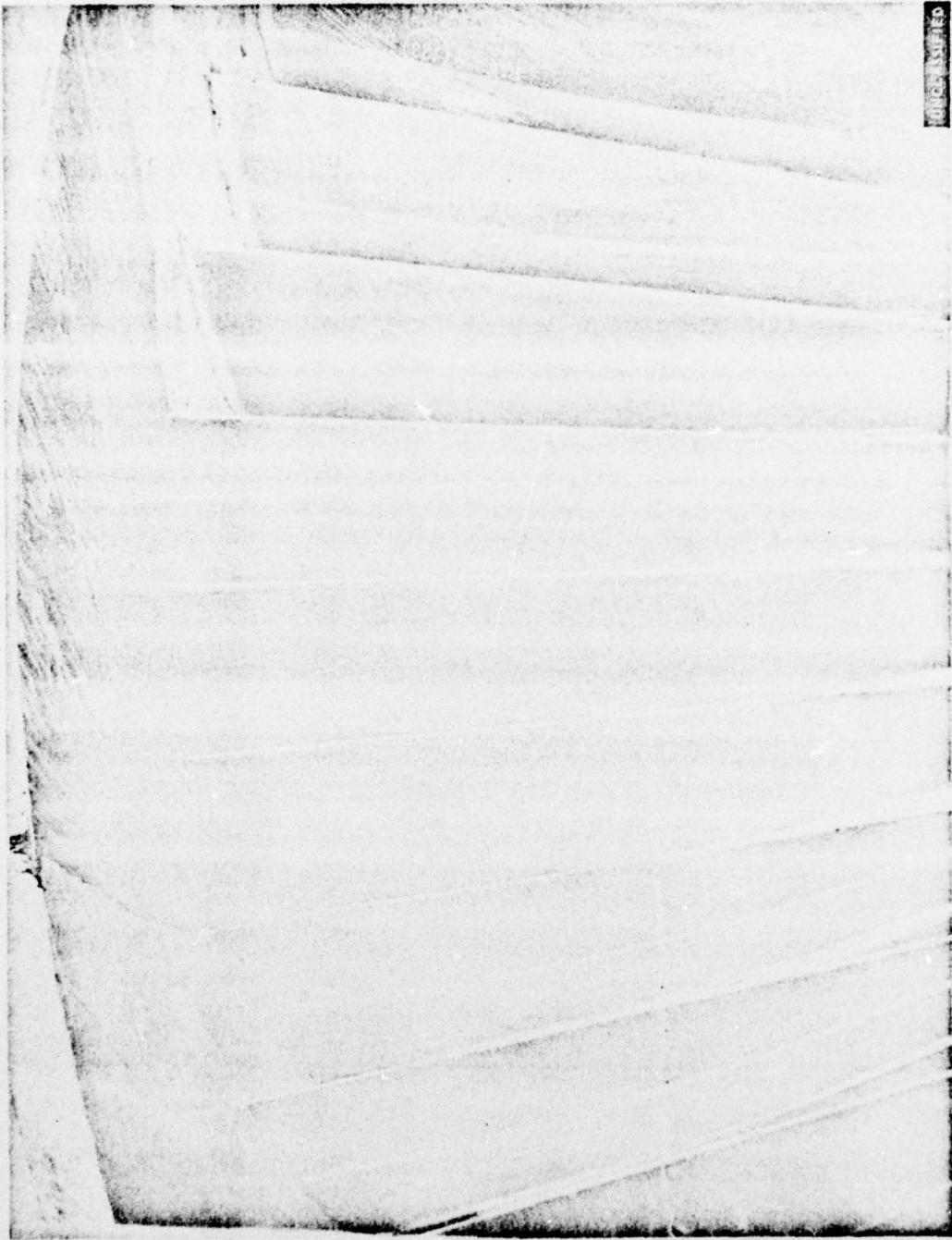


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Figure 195 Oil and Graphite Flow Patterns, Second Vane Recambering Design A Airfoil  
at -6 Degrees Incidence. Midspan Exit Mach No. = 0.869 (FE95863)

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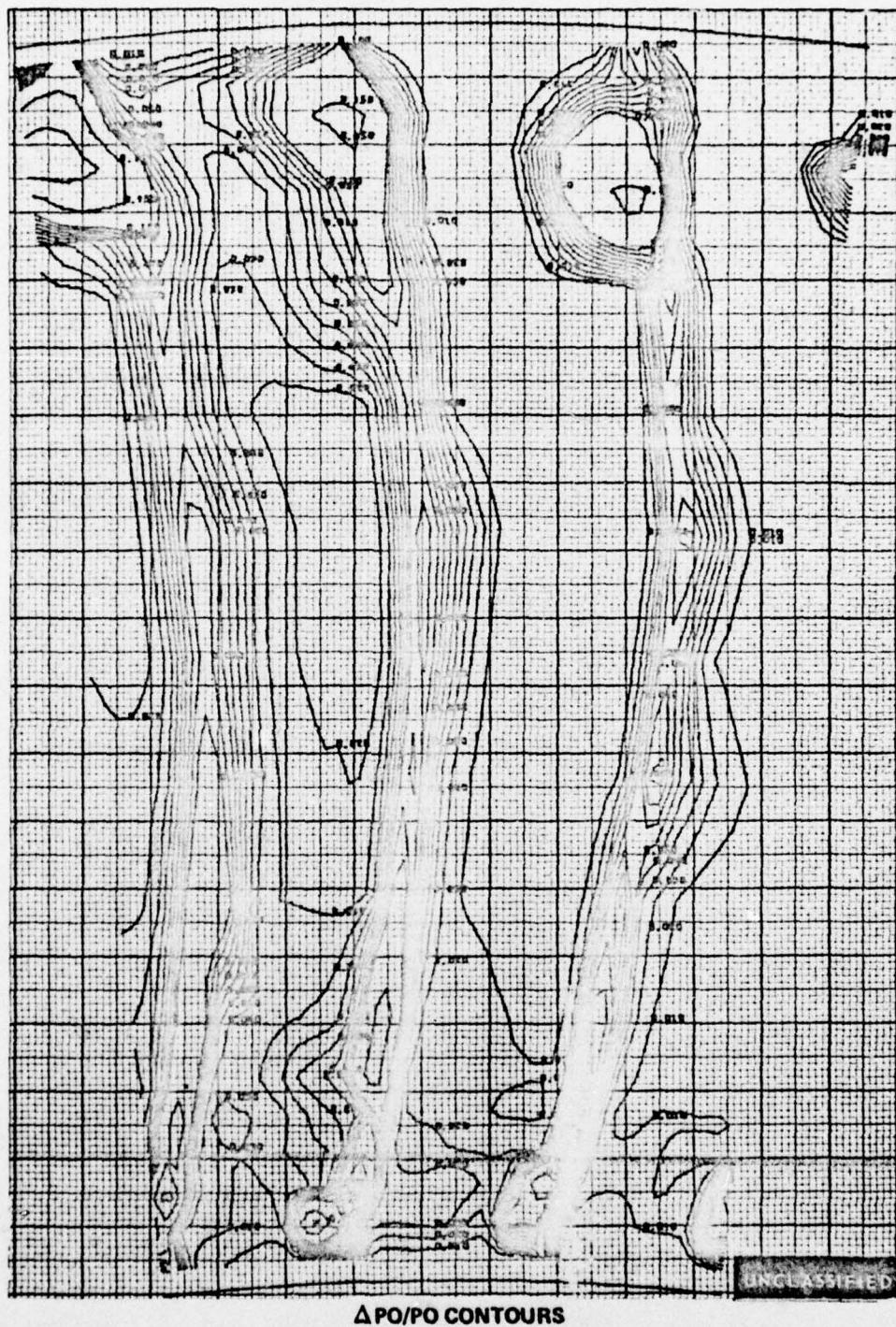
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**Figure 196** Oil and Graphite Flow Patterns, Second Vane Recambering Design A Airfoil  
at -6 Degrees Incidence. Midspan Exit Mach No. = 0.869 (FE95864)

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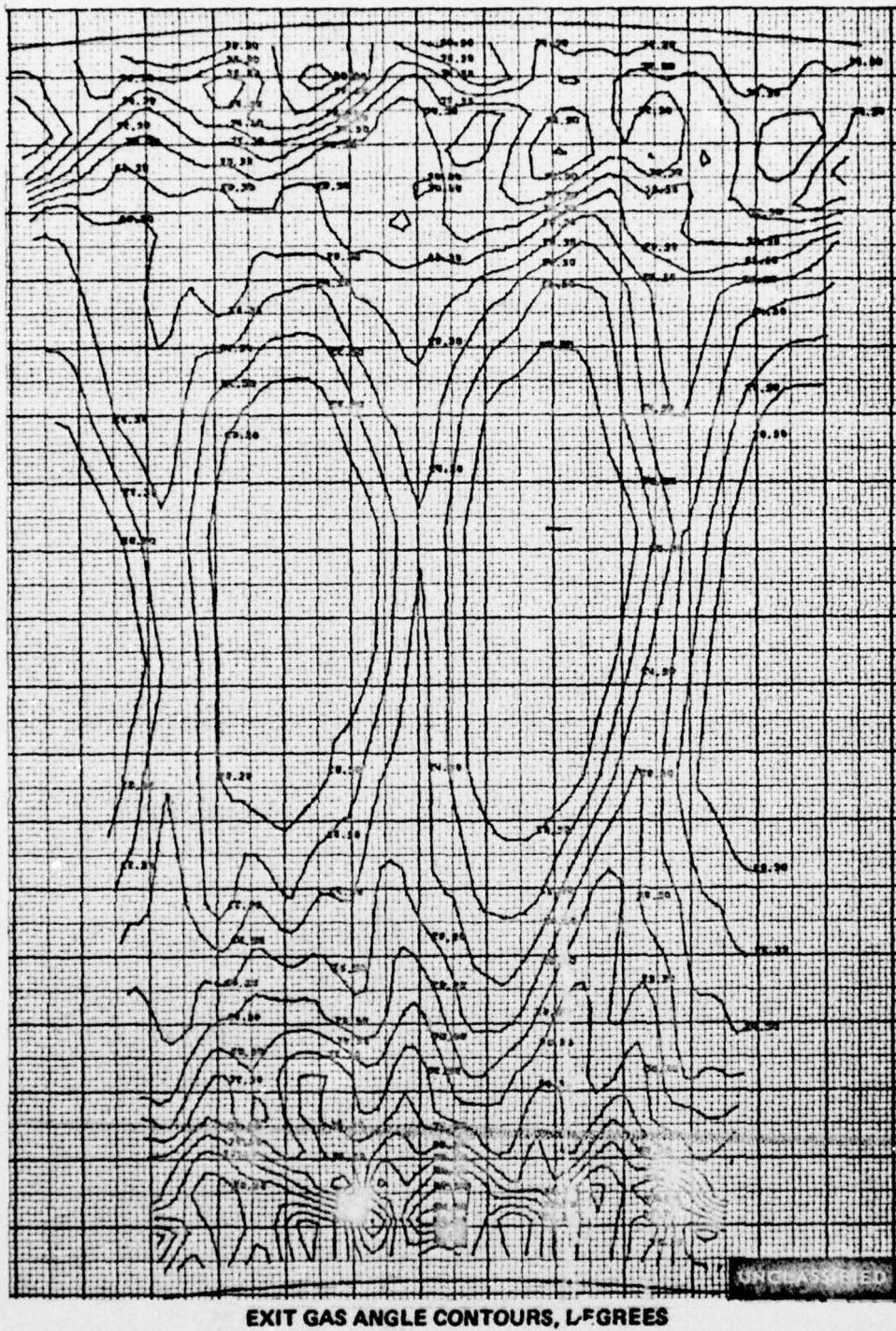
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**Figure 197 Pressure Loss Contours, Second Vane Recambering Design C at +10° Incidence. Three Flow Passages. Midspan Exit Mach Number = 0.858**

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**Figure 198** Exit Gas Angle Contours, Second Vane Recambering Design C at +10° Incidence. Three Flow Passages. Midspan Exit Mach Number = 0.858

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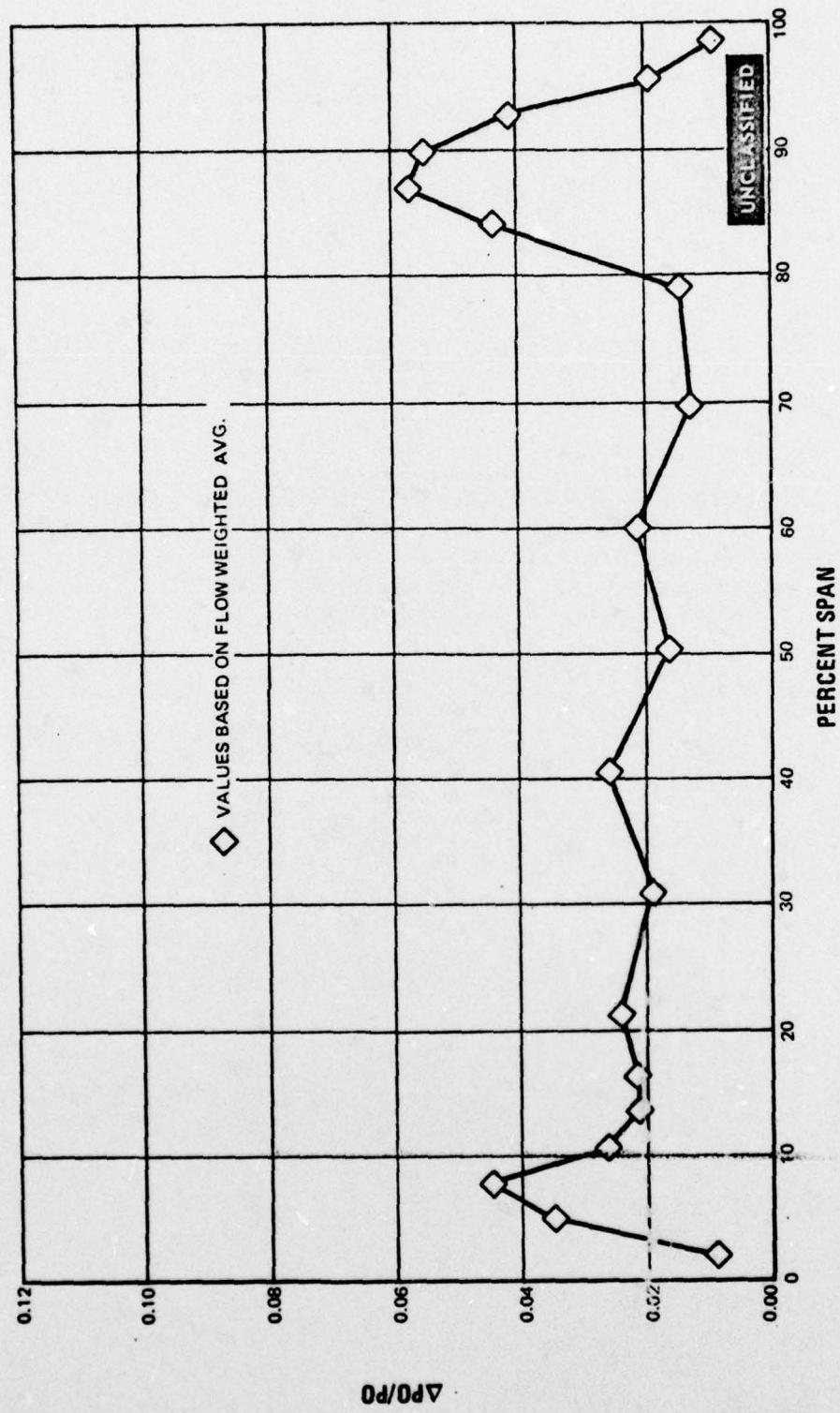


Figure 199 Spanwise Pressure Loss Distribution, Second Vane Recambering Design C  
at +10° Incidence. Midspan Exit Mach Number = 0.858

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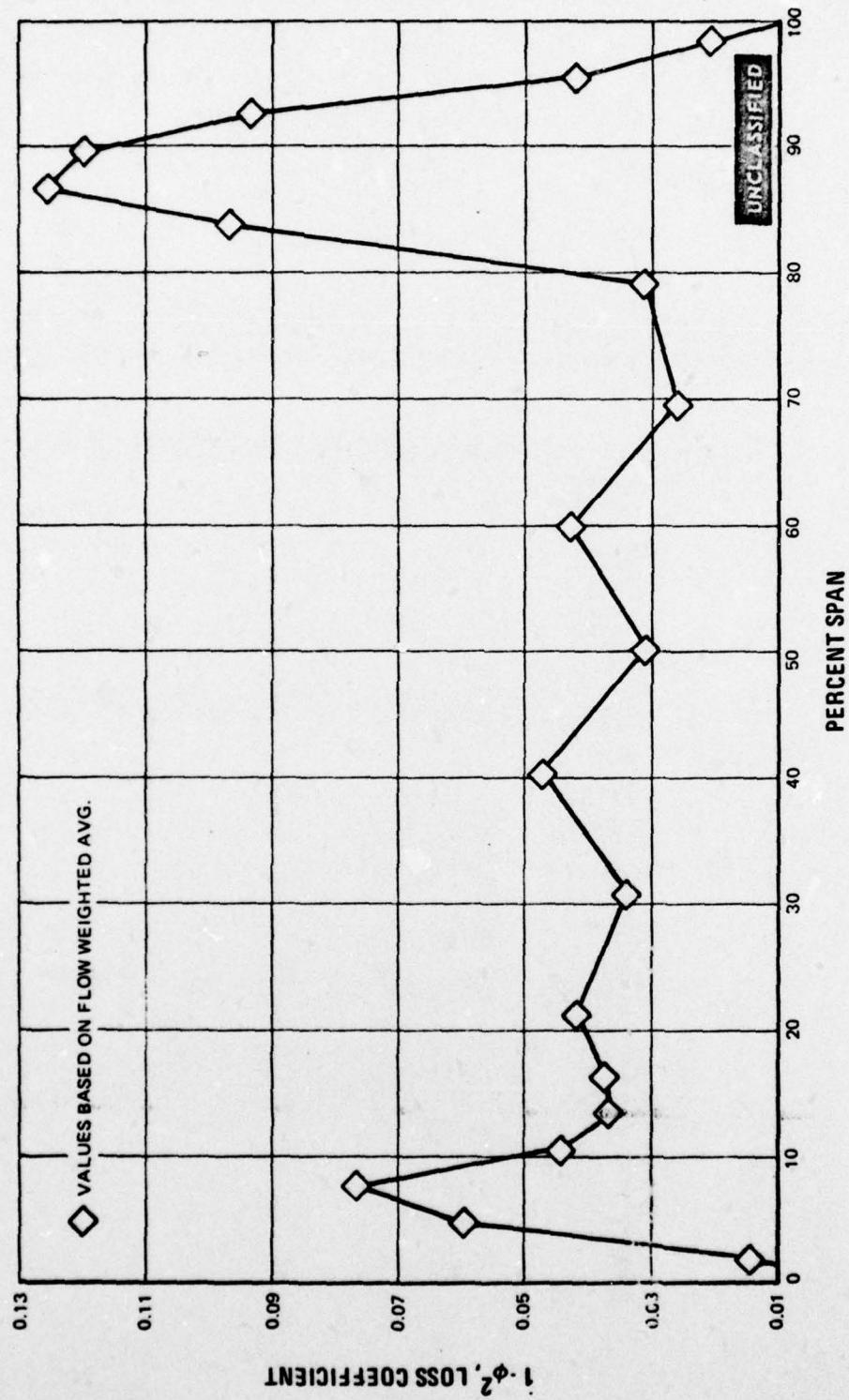


Figure 200 Spanwise Loss Coefficient Distribution, Second Vane Recambering Design C  
at +10° Incidence. Midspan Exit Mach Number = 0.858

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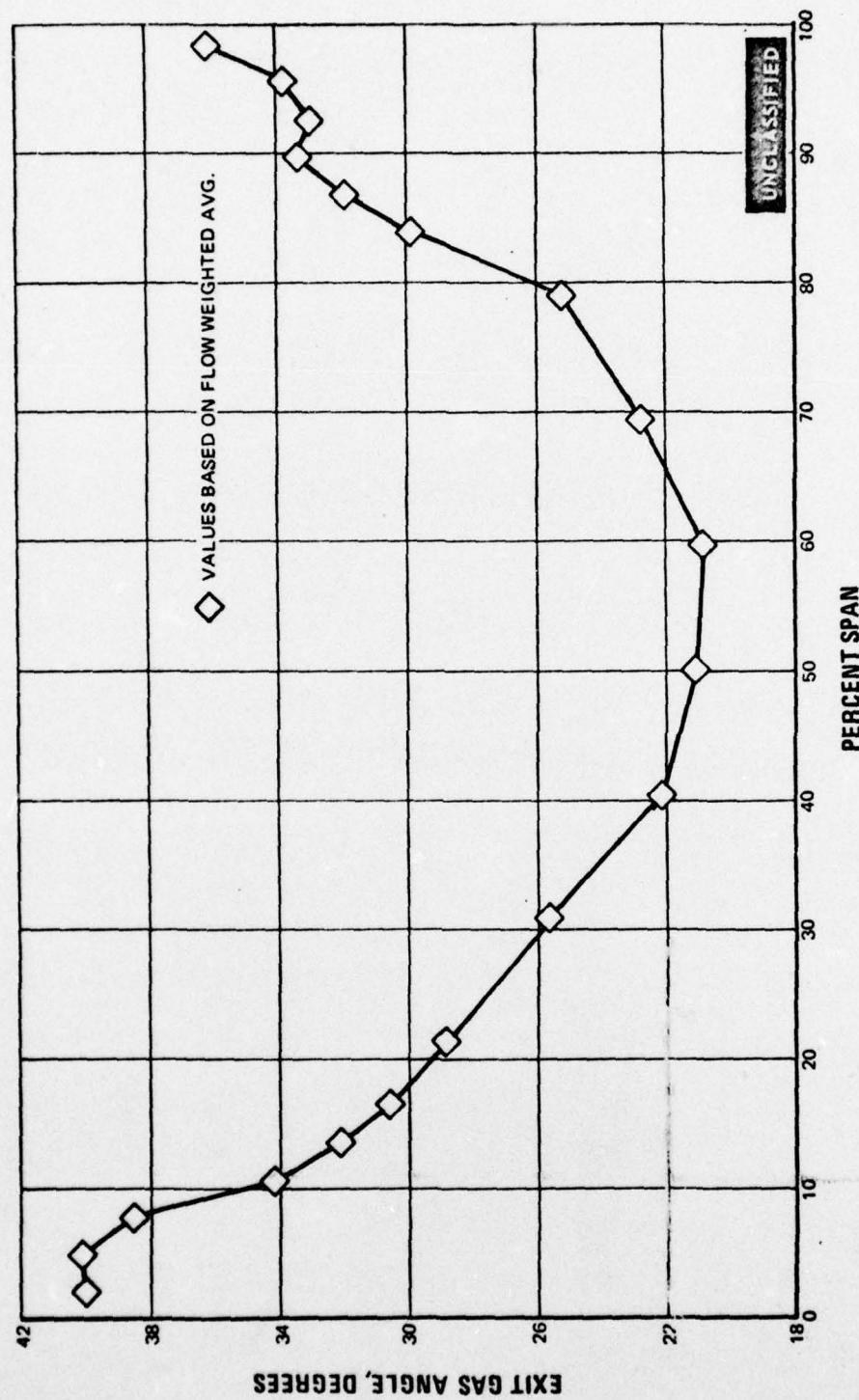


Figure 201 Spanwise Exit Gas Angle Distribution, Second Vane Recambering Design C  
at +10° Incidence. Midspan Exit Mach Number = 0.858

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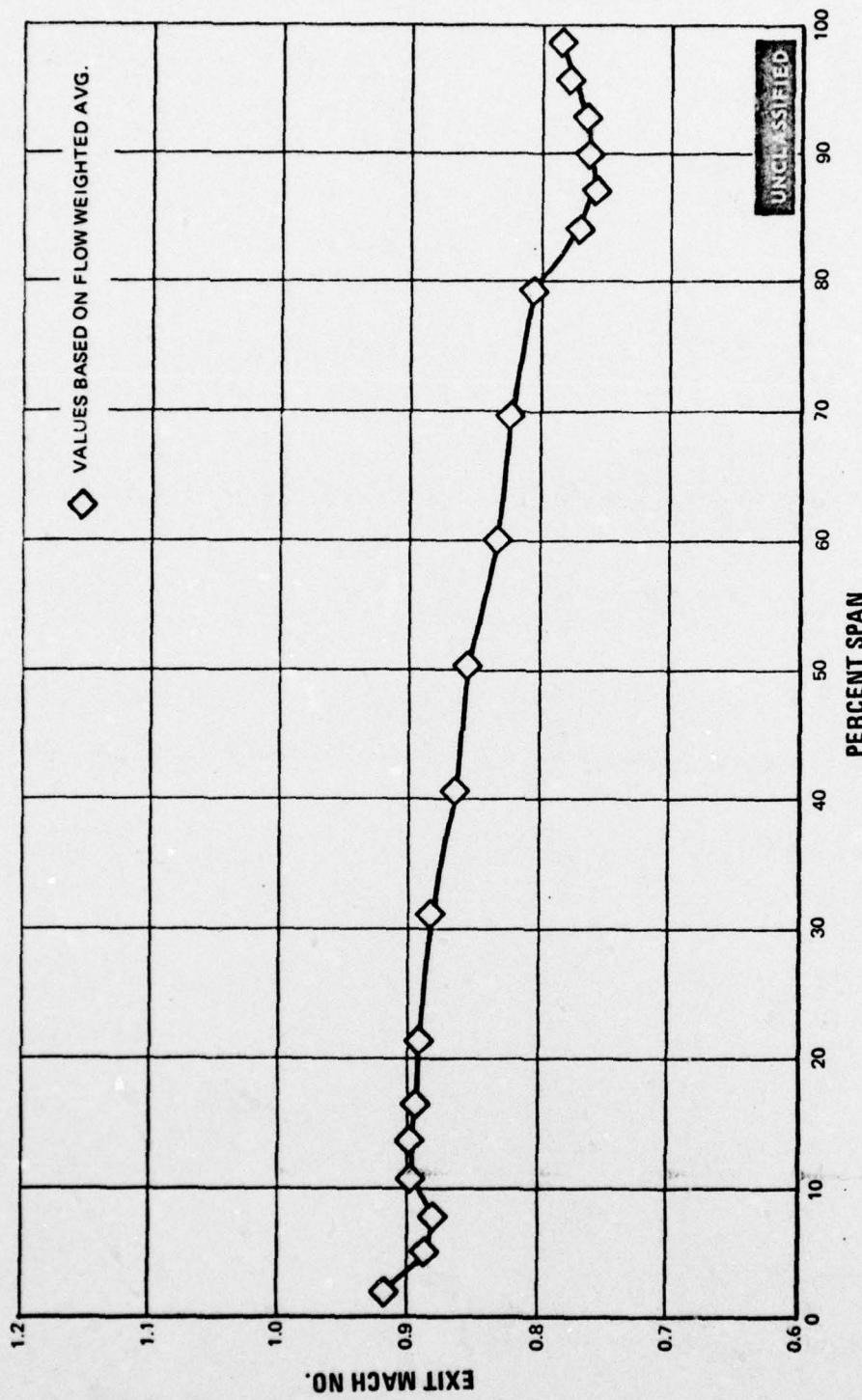


Figure 202 Spanwise Exit Mach Number Distribution, Second Vane Recambering Design  
C at +10° Incidence. Midspan Exit Mach Number = 0.858

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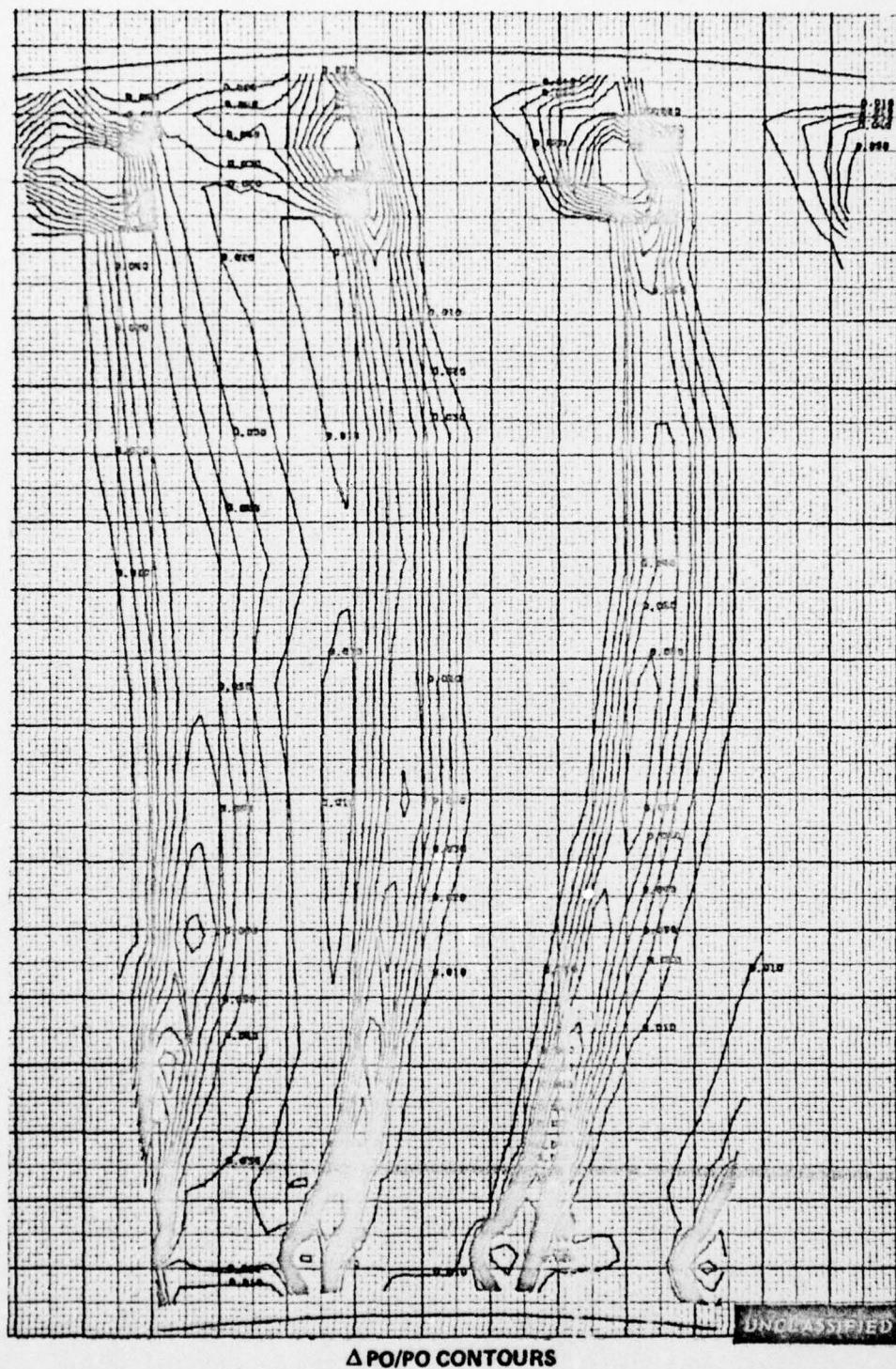
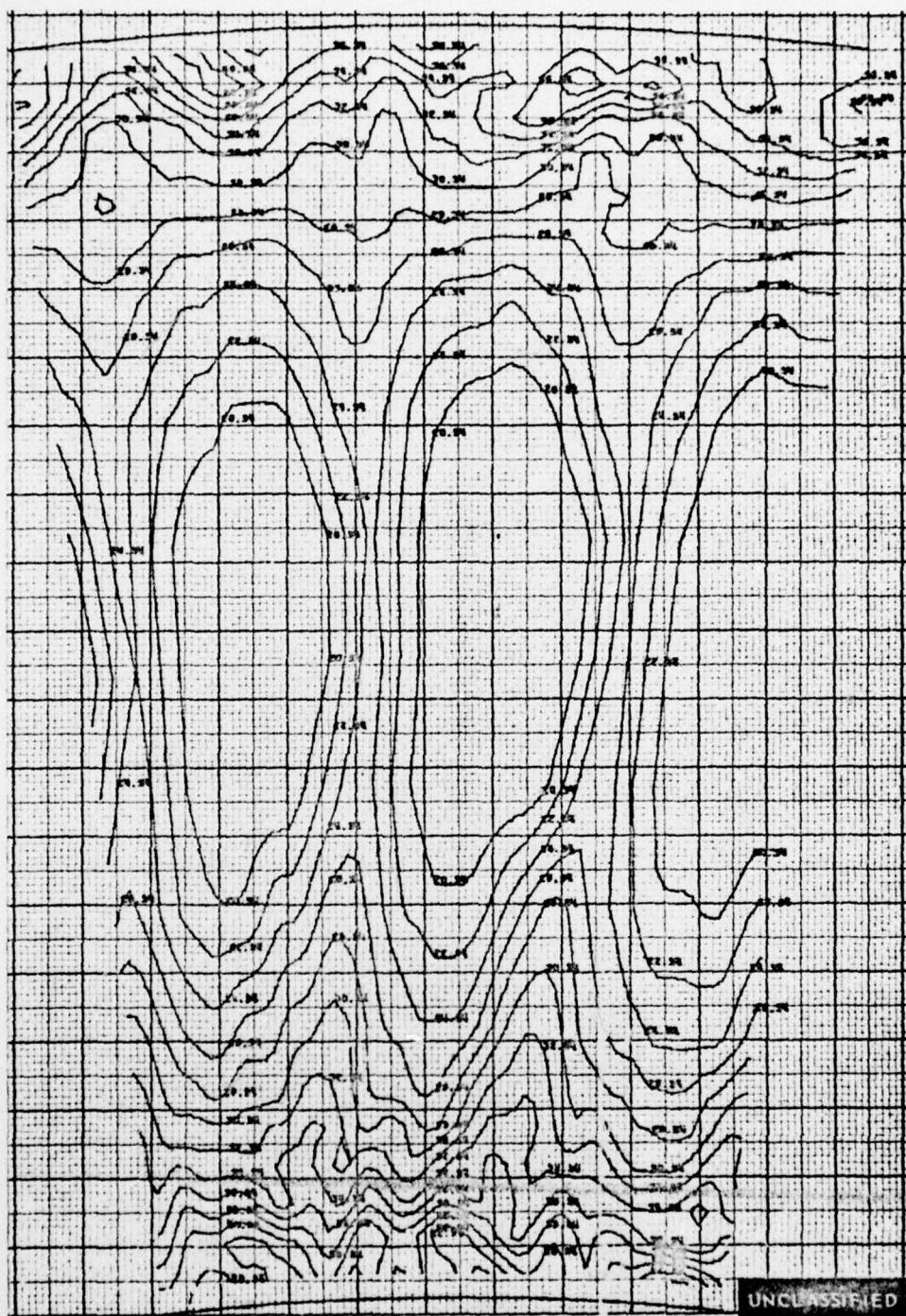


Figure 203 Pressure Loss Contours, Second Vane Recombining Design C at  $-6^\circ$  Incidence.  
Three Flow Passages. Midspan Exit Mach Number = 0.879

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EXIT GAS ANGLE CONTOURS, DEGREES

Figure 204 Exit Gas Angle Contours, Second Vane Recambering Design C at -6° Incidence. Three Flow Passages. Midspan Exit Mach Number = 0.879

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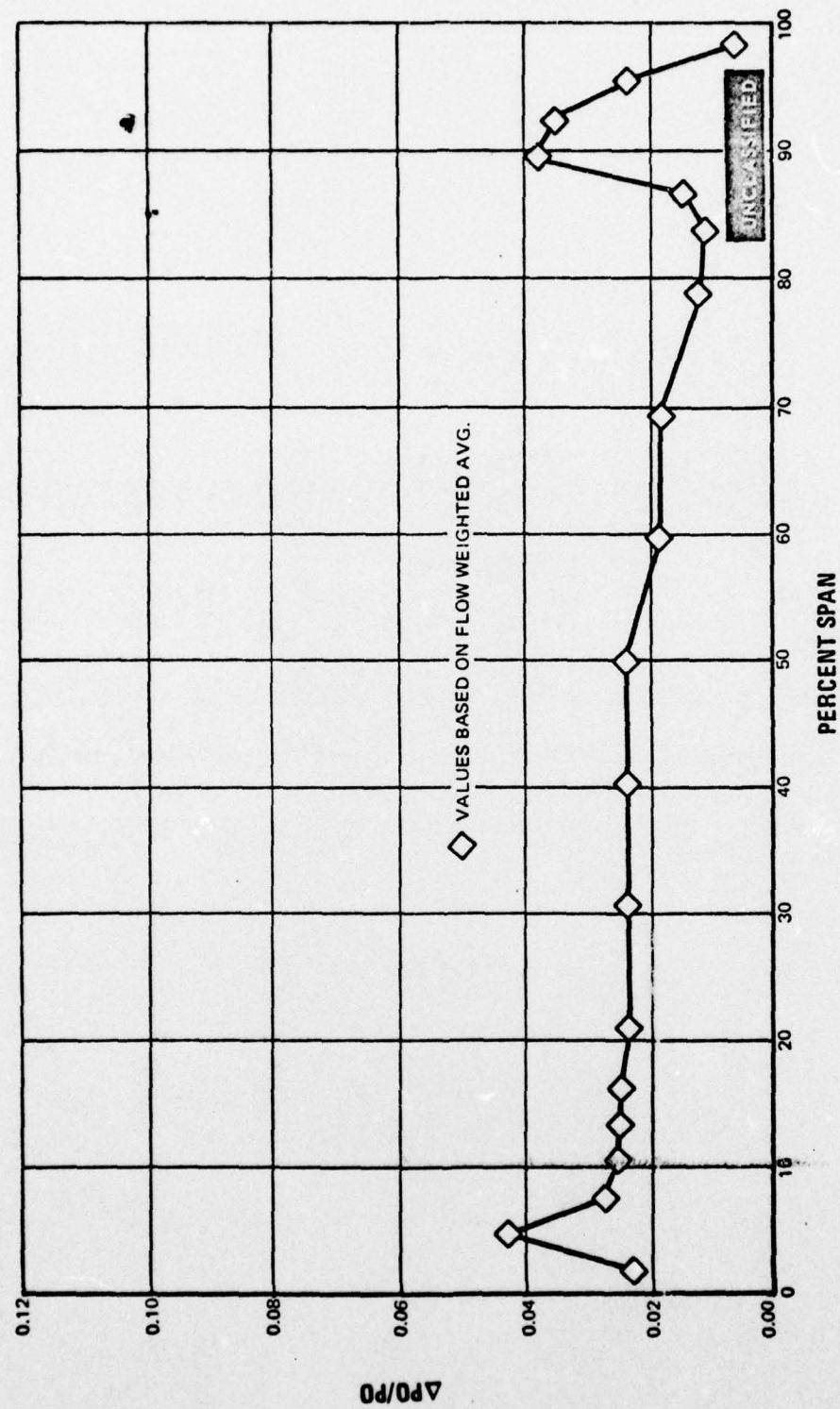


Figure 205 Spanwise Pressure Loss Distribution, Second Vane Recambering Design C  
at -6° Incidence. Midspan Exit Mach Number = 0.879

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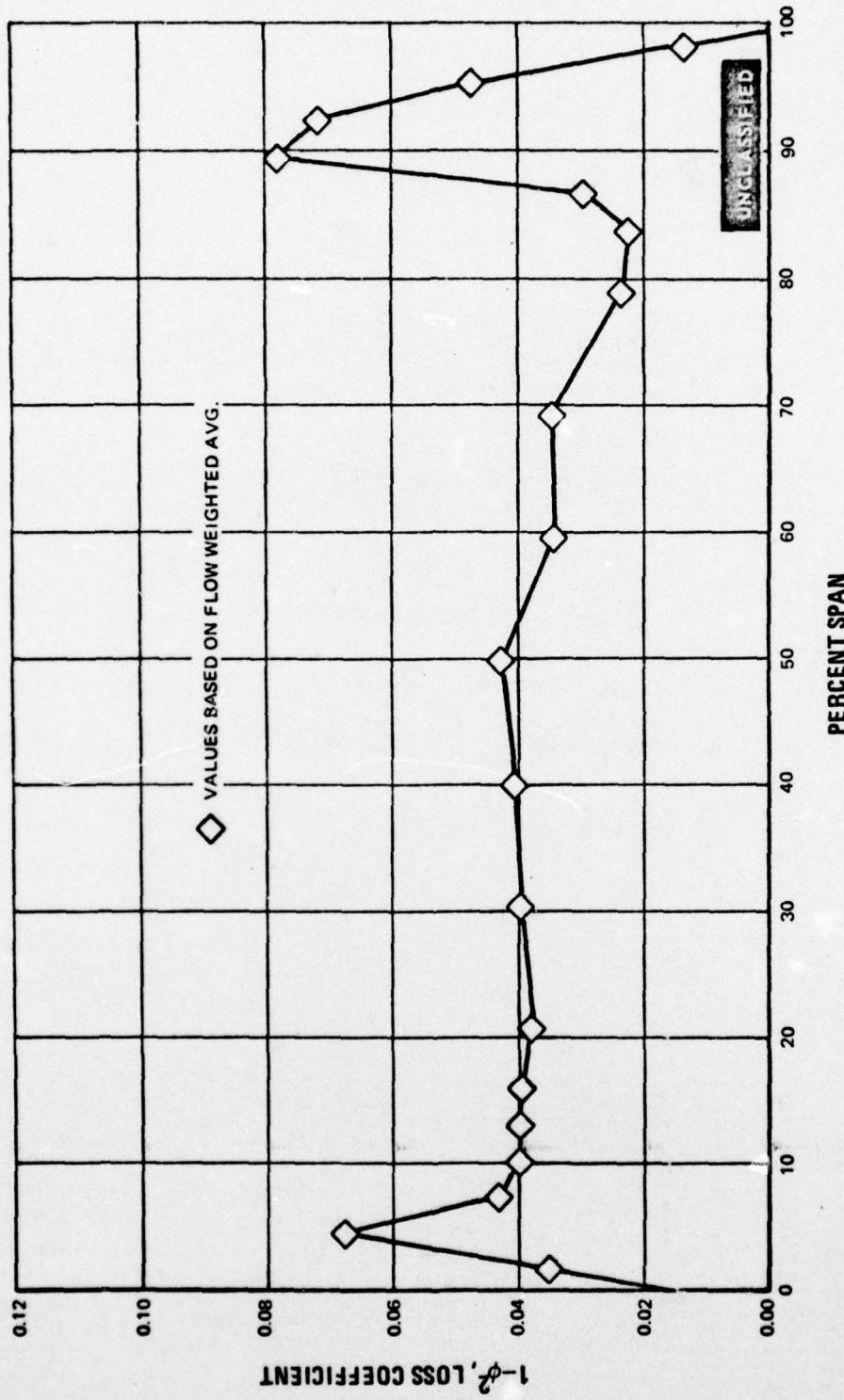


Figure 206 Spanwise Loss Coefficient Distribution, Second Vane Recambering Design C  
at  $-6^\circ$  Incidence. Midspan Exit Mach Number = 0.879

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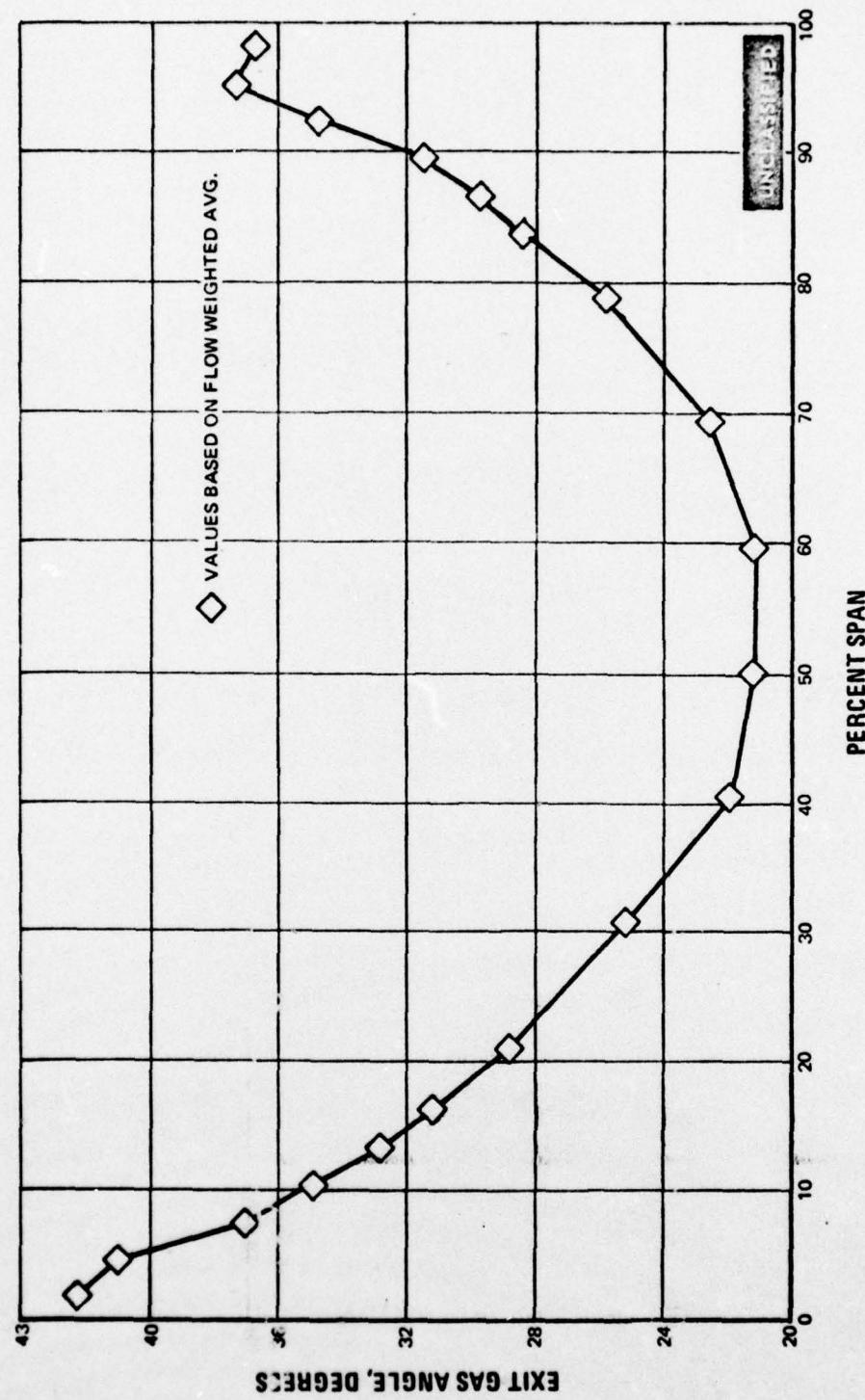


Figure 207 Spanwise Exit Gas Angle Distribution, Second Vane Recombining Design C  
at -6° Incidence. Midspan Exit Mach Number = 0.879

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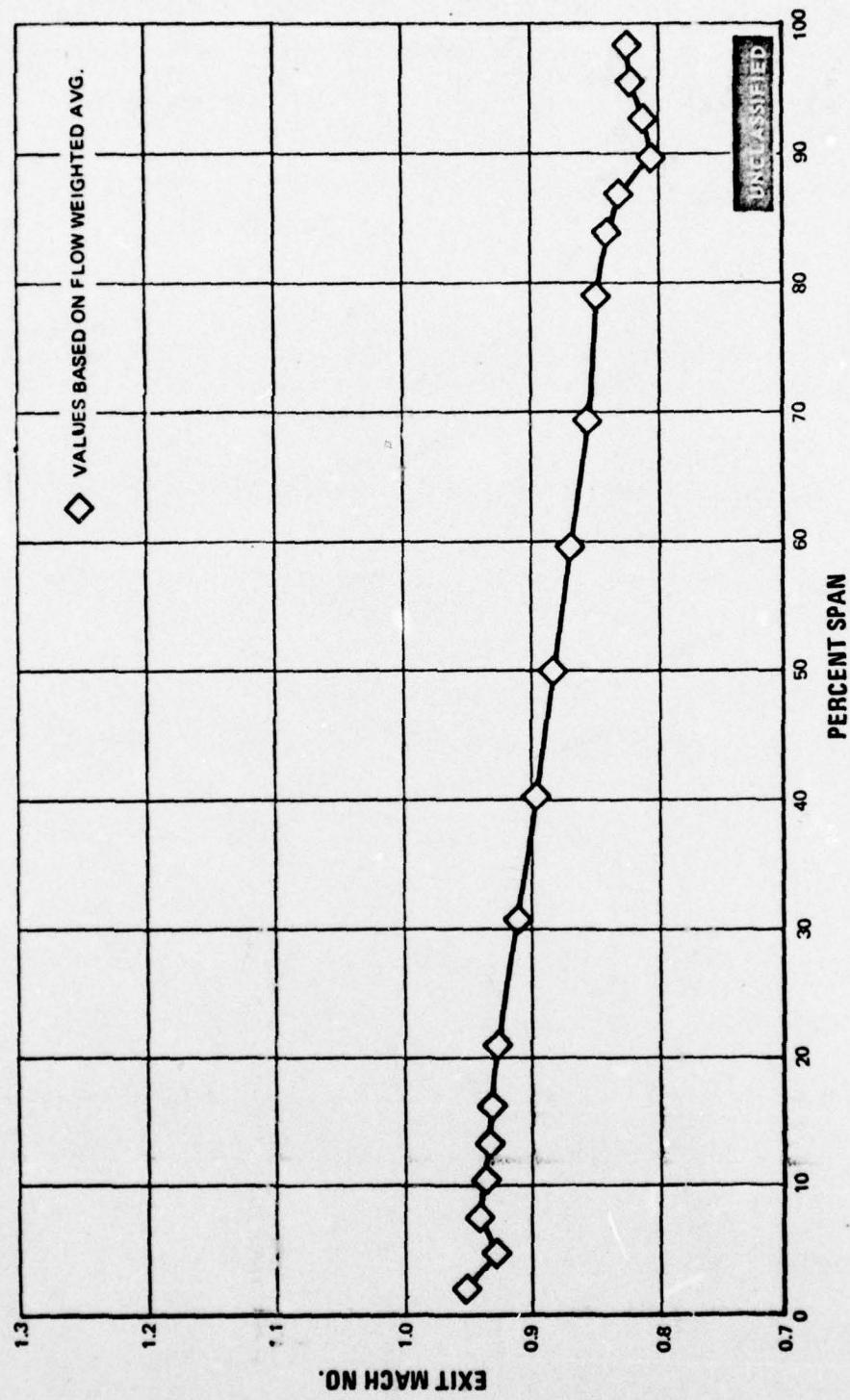


Figure 208 Spanwise Exit Mach Number Distribution, Second Vane Recambering Design  
C at -6° Incidence. Midspan Exit Mach Number = 0.879

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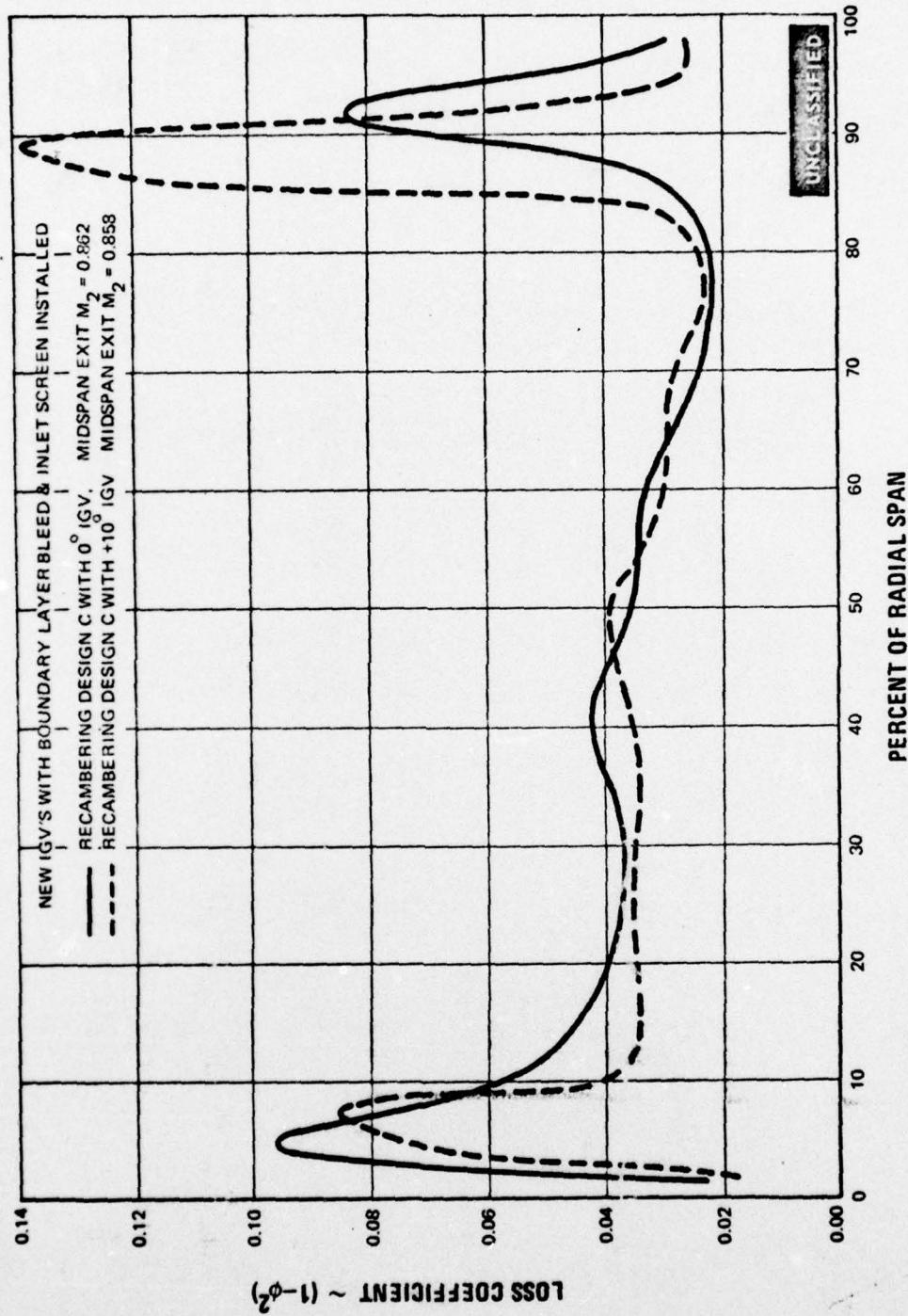
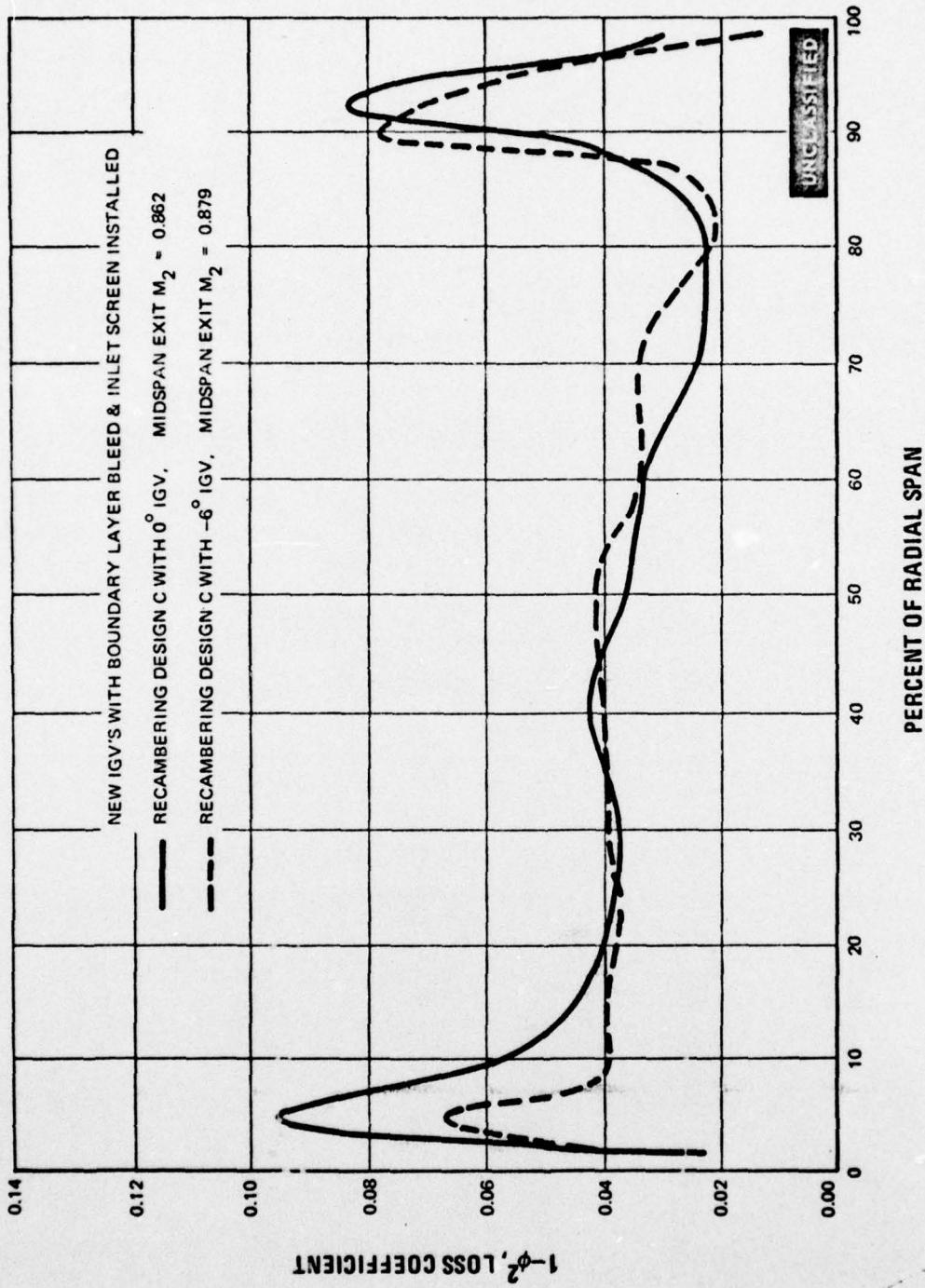


Figure 209 Comparison of Recambering Design C Airfoil Loss Coefficient Distribution at +10 Degrees Incidence With Zero Incidence Values

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**Figure 210** Comparison of Recambering Design C Airfoil Loss Coefficient Distribution at -6 Degrees Incidence With Zero Incidence Values

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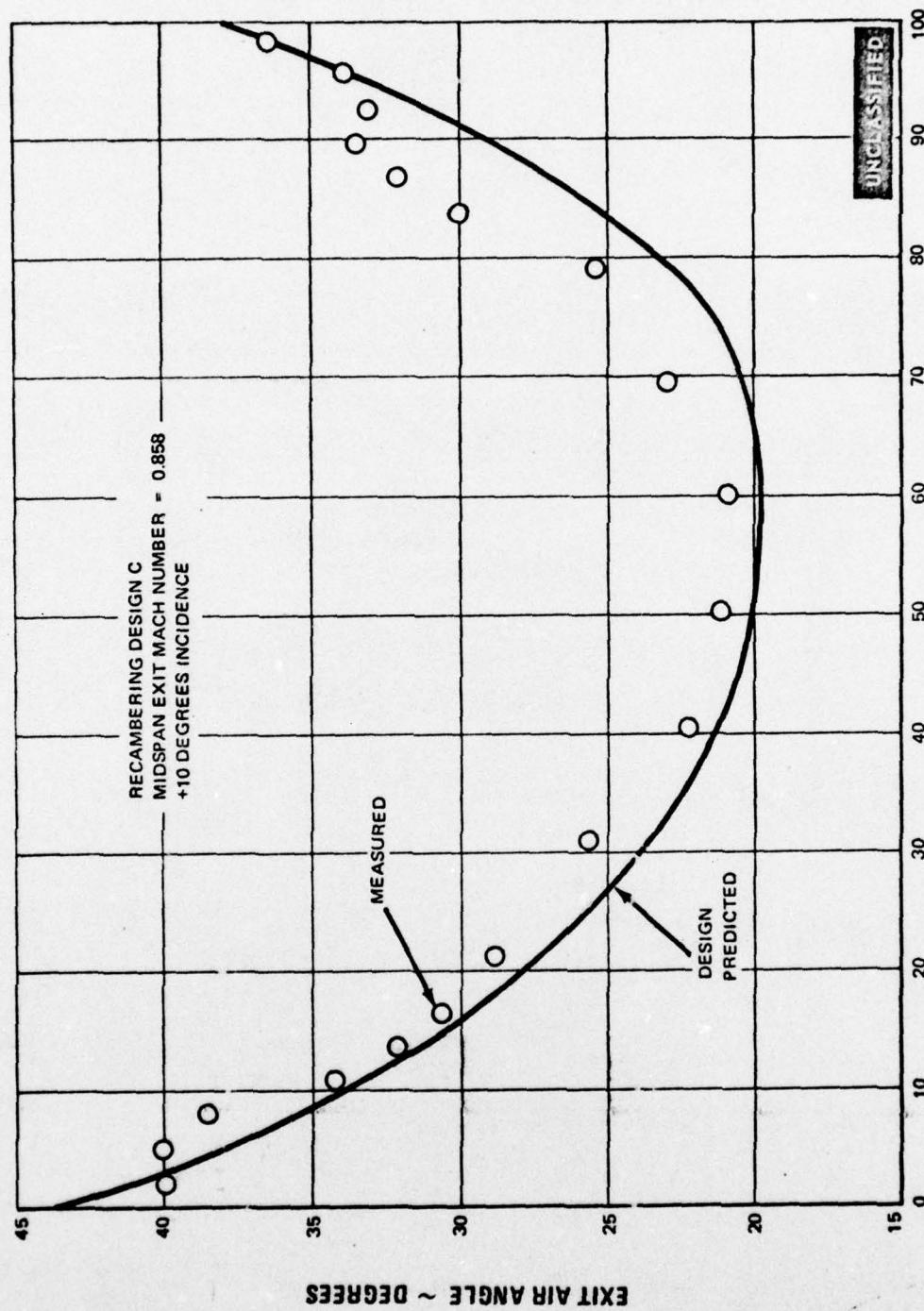


Figure 211 Comparison of Measured Exit Gas Angle With Predicted Design Values at +10 Degrees Incidence. Design C Airfoil at Midspan Exit Mach No. = 0.858

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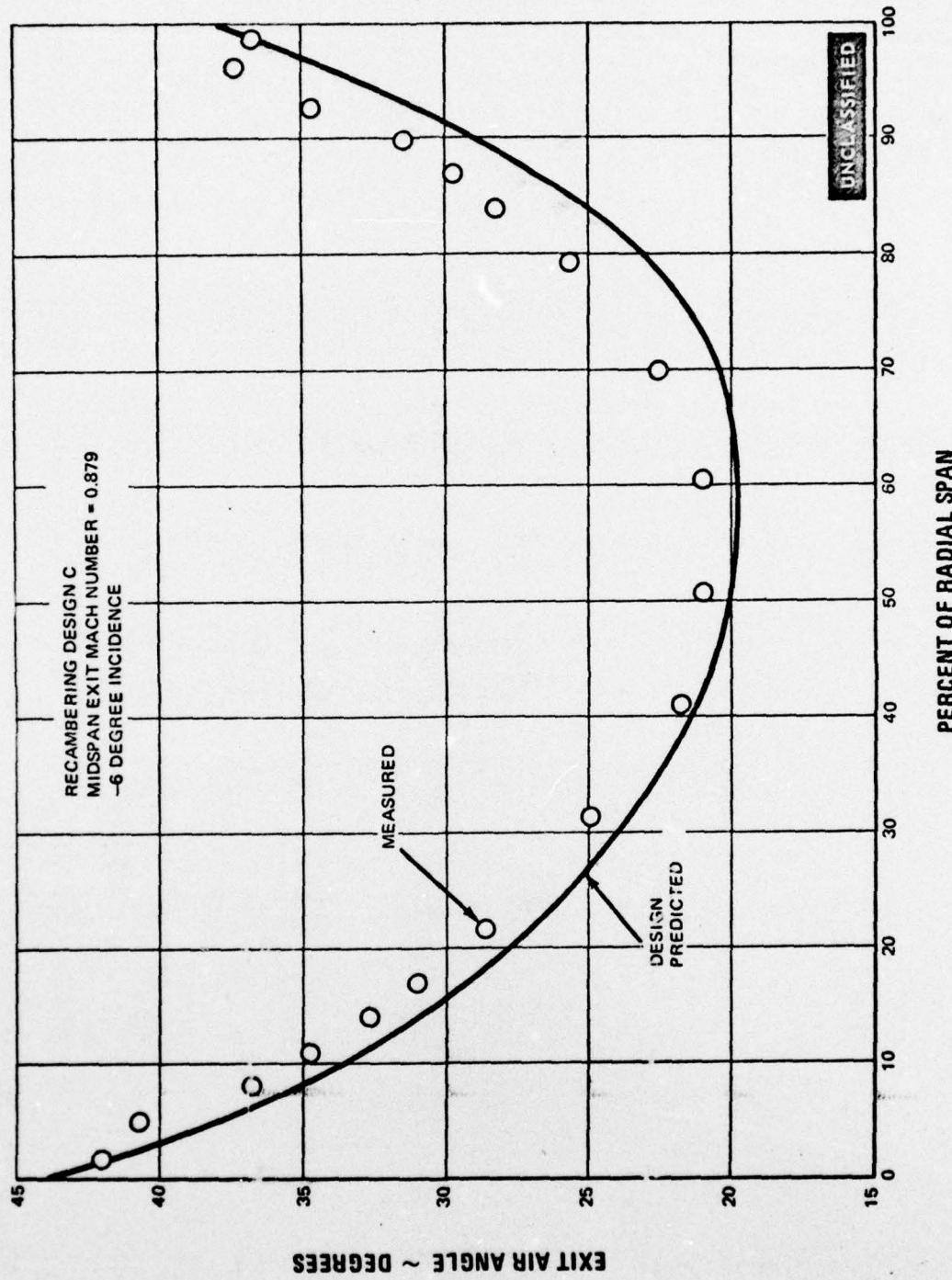


Figure 212 Comparison of Measured Exit Gas Angle With Predicted Design Values at -6 Degrees Incidence. Design C Airfoil at Midspan Exit Mach No. = 0.879

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**Figure 213** Oil and Graphite Flow Patterns, Second Vane Recambering Design C Airfoil at +10 Degrees Incidence. Midspan Exit Mach No. = 0.858 (FE95749)

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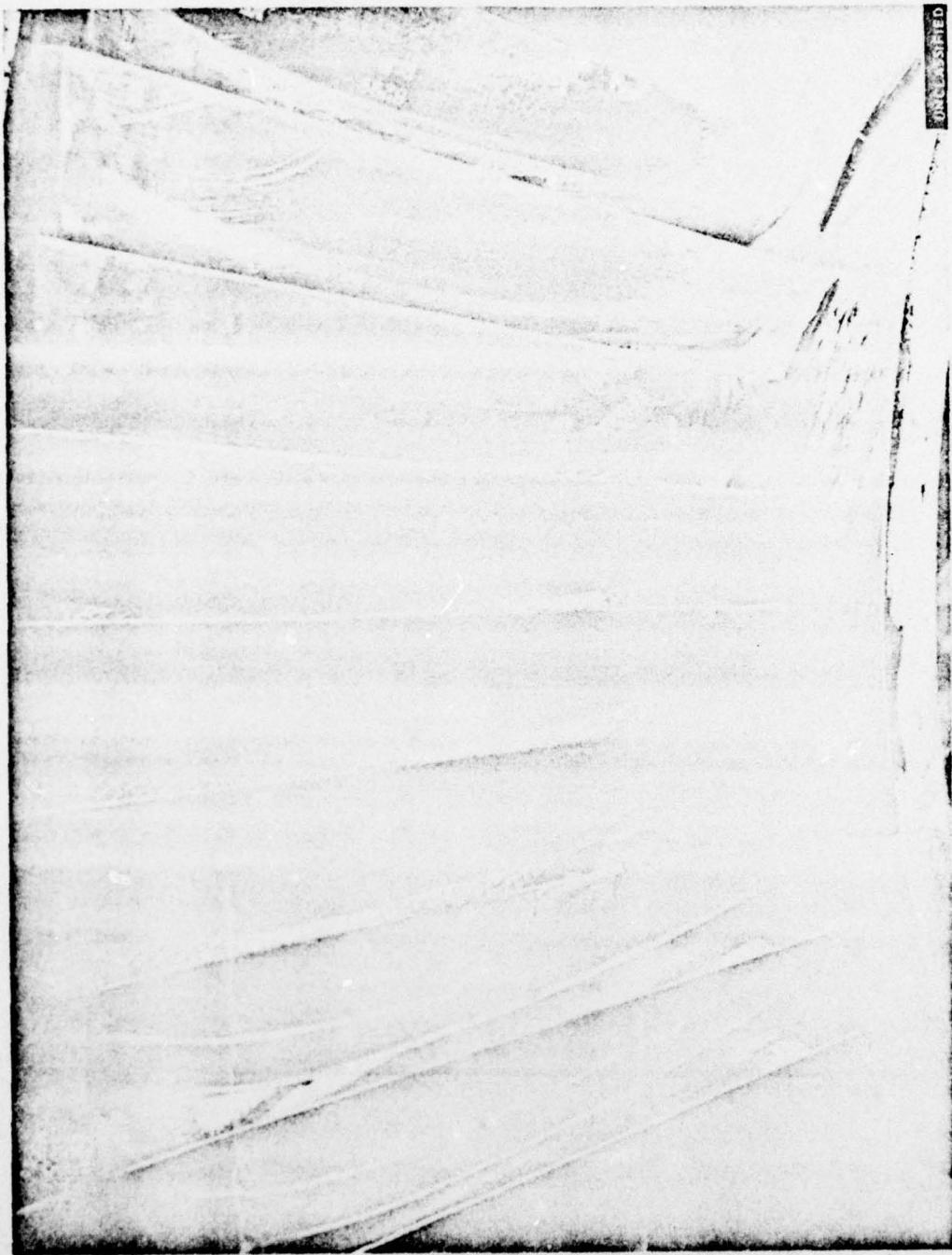


Figure 214 Oil and Graphite Flow Patterns, Second Vane Recambering Design C Airfoil  
at +10 Degrees Incidence. Midspan Exit Mach No. = 0.858 (FE95750)

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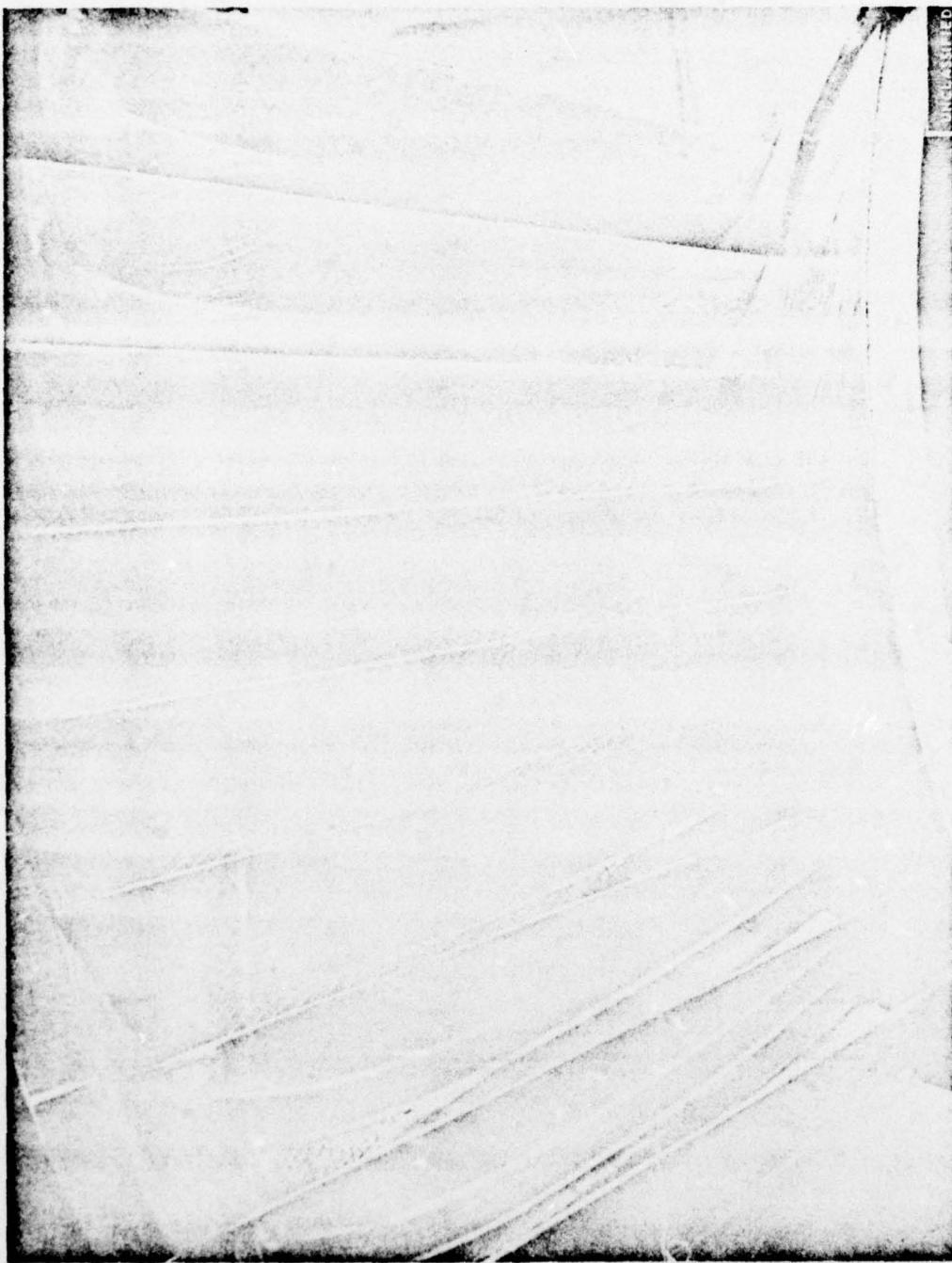


Figure 215 Oil and Graphite Flow Patterns, Second Vane Recambering Design C Airfoil  
at -6 Degrees Incidence. Midspan Exit Mach No. = 0.879 (FE95865)

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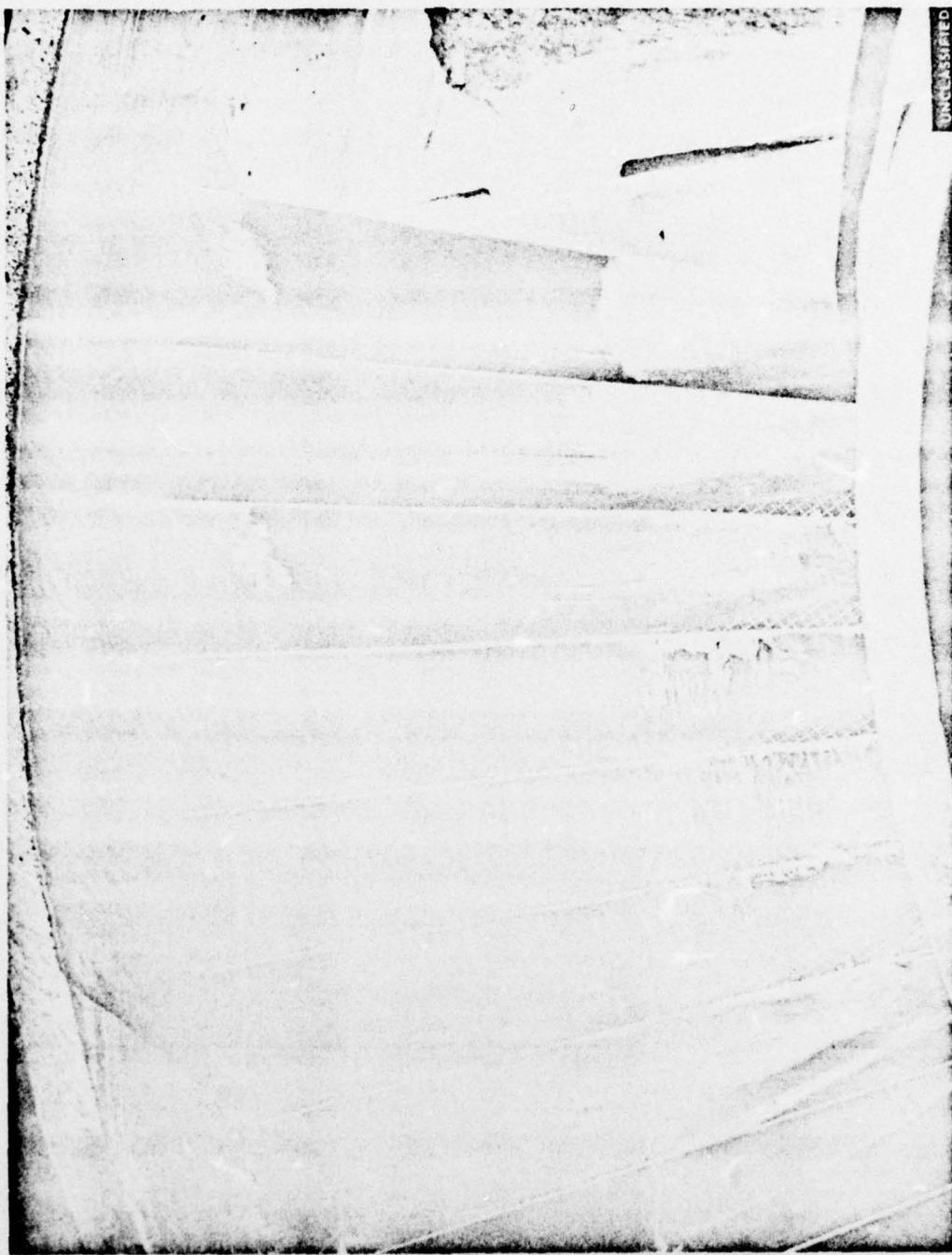


Figure 216 Oil and Graphite Flow Patterns, Second Vane Recambering Design C Airfoil  
at -6 Degrees Incidence. Midspan Exit Mach No. = 0.879 (FE95866)

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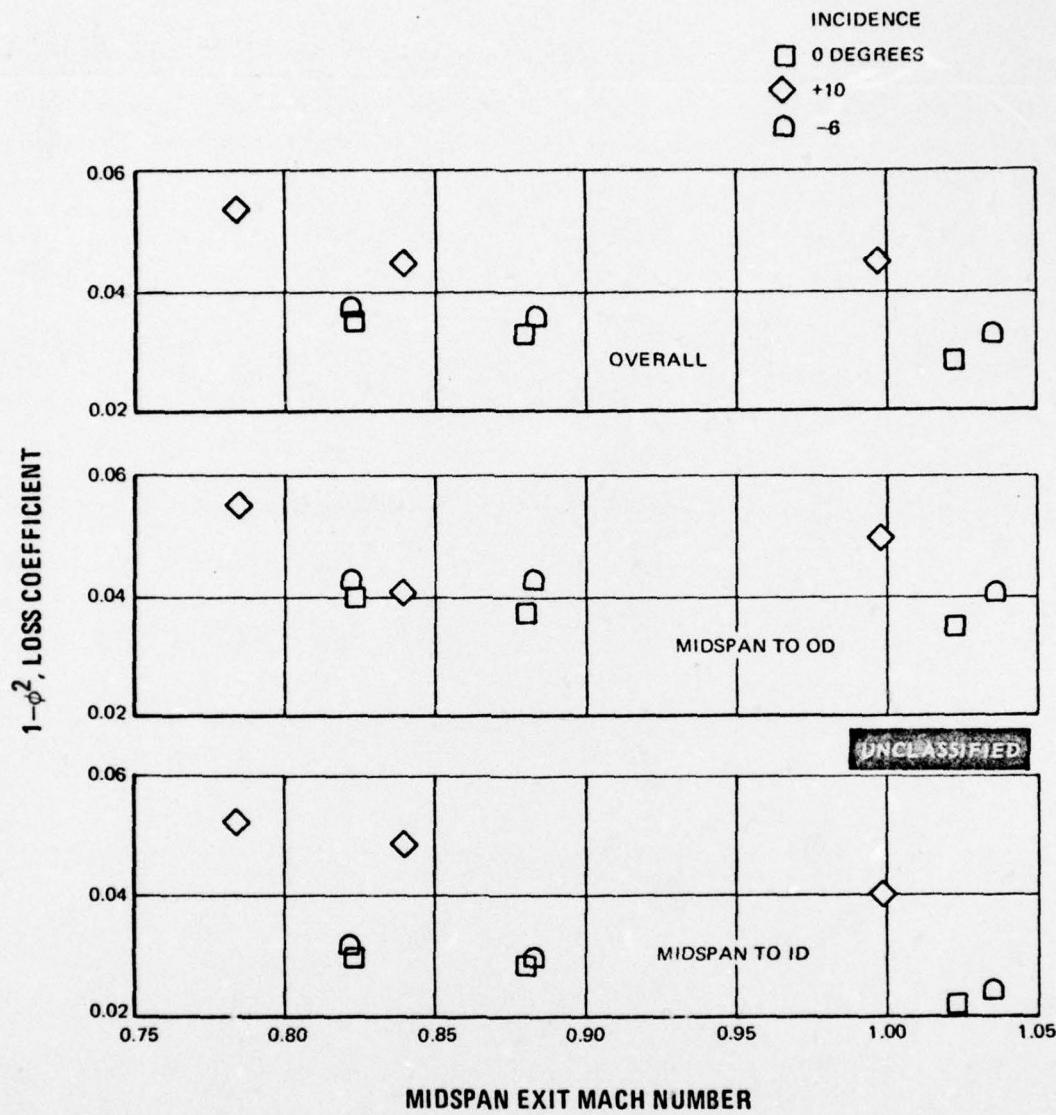


Figure 217 Effect of Mach Number on Loss for the First Recambered Airfoil at Various Incidence Angles

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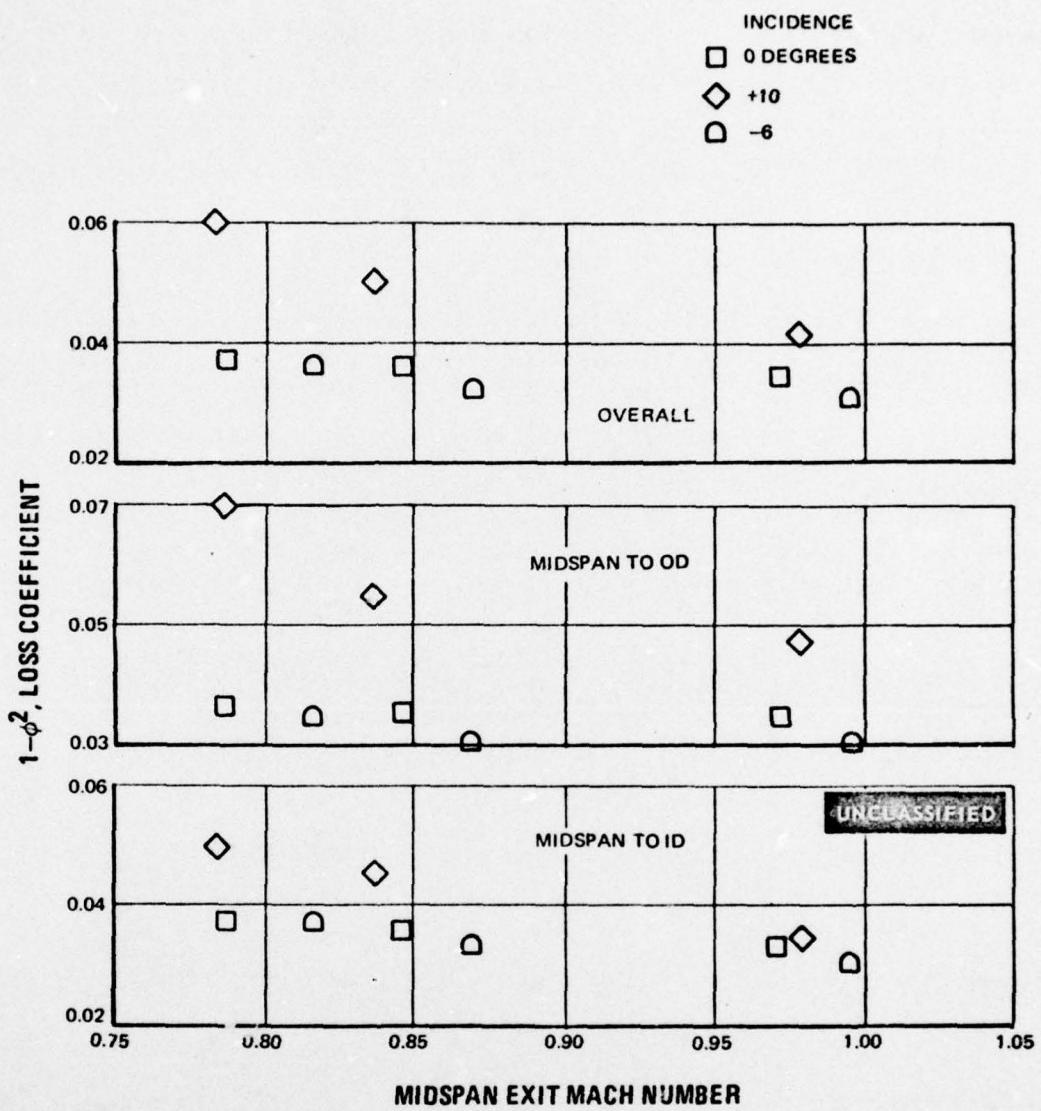


Figure 218 Effect of Mach Number on Loss for Recamembering Design A at Various Incidence Angles

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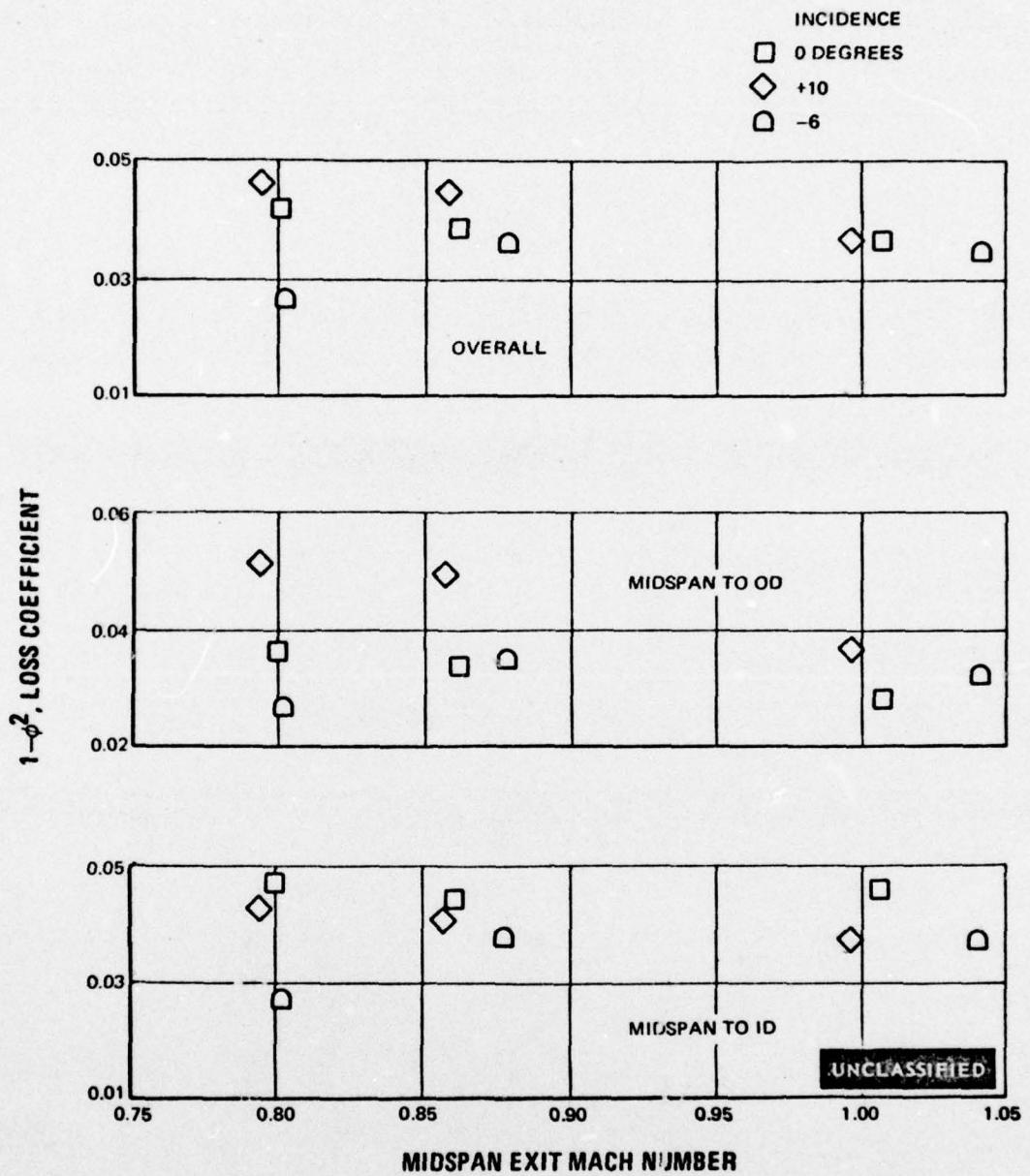


Figure 219 Effect of Mach Number on Loss for Recambering Design C at Various Incidence Angles

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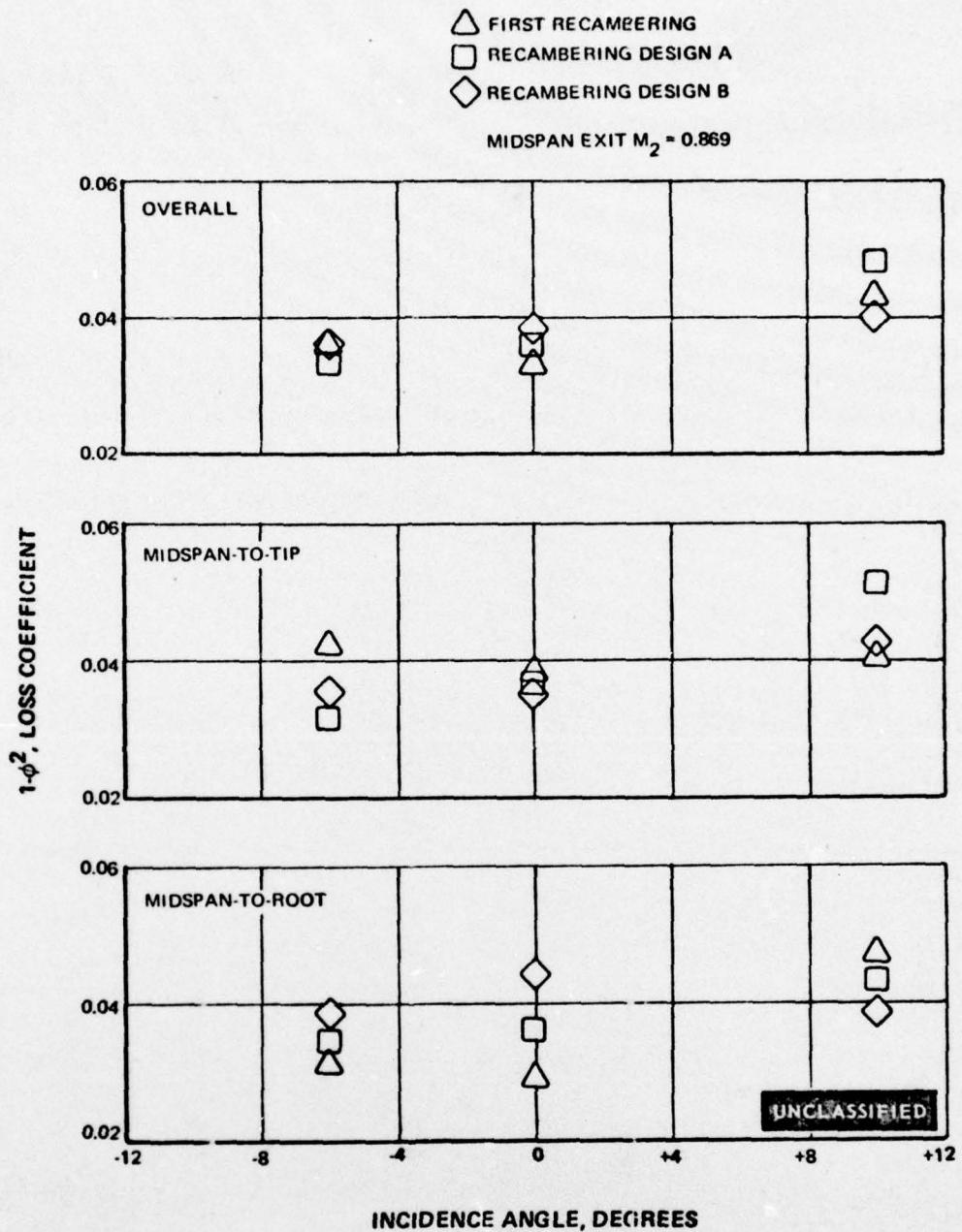


Figure 220 Effect of Incidence on Loss for the Various Recambered Second Vane Airfoils.  
Midspan Exit Mach Number = 0.869

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## 5. SUMMARY

(U) The results of the off-design evaluation of the second vane baseline and recambered designs are summarized in Table XXXIII. The values shown in this table are those at the design midspan exit Mach number of 0.869 and were obtained by interpolation from Figures 217 through 219.

TABLE XXXIII  
SECOND VANE RECAMBERING LOSS COEFFICIENTS  
CORRECTED TO DESIGN POINT MACH NUMBER = 0.869

Incidence, Degrees	Midspan-To-Root			Midspan-To-Tip			Overall		
	+10	0	-6	+10	0	-6	+10	0	-6
Baseline	0.039	0.035	0.029	0.035	0.035	0.029	0.038	0.035	0.029
First Recambering	0.047	0.029	0.031	0.040	0.038	0.042	0.043	0.033	0.036
Recambering Design A	0.043	0.036	0.034	0.051	0.035	0.031	0.048	0.036	0.033
Recambering Design B	—	0.031	—	—	0.033	—	—	0.032	—
Recambering Design C	0.038	0.044	0.038	0.042	0.034	0.035	0.040	0.038	0.036

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(U) The following results are noted:

- For the baseline airfoil, the overall integrated loss level and the integrated loss coefficients at the end-wall regions have increased for +10 degrees incidence and decreased for -6 degrees incidence, compared to the zero incidence levels.
- For the first recambered airfoil, the overall integrated losses are higher for both the +10 degrees and -6 degrees incidence levels, compared to the zero incidence levels. At the +10 degrees incidence, the integrated losses are higher than the zero incidence condition at both the midspan and end-wall regions. At -6 degree incidence, the losses are relatively unchanged over the airfoil span except in the tip region, where the -6 degree incidence losses are significantly higher.

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- For the airfoil with recambering design A, integrated losses were significantly higher at the +10 degrees incidence level than at zero incidence, and the losses at the -6 degrees incidence were essentially unchanged, compared to the zero incidence condition.
- For the airfoil with recambering design C, the overall integrated loss level was higher at +10 degrees incidence and lower at -6 degrees incidence, compared to zero incidence. In the root region, the losses at +10 degree incidence appeared to be slightly less than the zero incidence data. At -6 degrees incidence, the loss level is significantly below the zero incidence results near the root.

(U) The following conclusions are drawn from these results:

- The recambered airfoils generally showed an increase in loss coefficient with incidence (-6 to +10 degrees) which indicates that the airfoils are close to their optimum incidence
- None of the airfoils tested were sensitive to angle of attack, and the flow patterns of the airfoils did not change with incidence angle
- The tested airfoils did not show the transonic drag rise measured in the plane cascade evaluation (Section III). The difference may be due to the differences in the turbulence levels of the two test rigs, and the fact that the annular segment cascade permits the flow a greater latitude for radial readjustment
- The annular segment cascade is more representative of flow conditions in a turbine than the plane cascade. Therefore, the transonic drag rise indicated in the plane cascade is not expected in the turbine
- The characteristics of the evaluated airfoils verified the soundness of the current turbine design system

## **6. TEST PROCEDURE**

(U) The test procedure employed during this Task was identical to that of the Task IIb baseline evaluation. This procedure was described in detail in Reference 3.

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## **SECTION VII**

### **DESIGN PROCEDURE MANUAL (TASK IIId)**

#### **1. RFP OBJECTIVE**

(U) Prepare the turbine design procedure manual for the high work turbine.

#### **2. TASK OBJECTIVE**

(U) The purpose of this Task is to prepare the Turbine Design Procedure Manual. This manual will contain detailed information for each computer program listed in the Contract proposal. This information will contain the following: a flow diagram; a listing of all input and output items and their definitions; a list of definitions for each term used in the computer code; a write-up of the pertinent equations; a listing of the computer code in Fortran IV; a copy of the computer program decks in Fortran IV; any necessary test cases for deck check-out.

(U) The loss relations presently used will be modified, if necessary, based on the results of the two- and three-dimensional testing during Phases II and III.

#### **3. STATUS**

(U) The following decks constitute the proposal design procedure manual:

- Turbine Meanline Design Program
- Turbine Stage Off-Design Program
- Turbine Streamline Analysis Program
- Airfoil Boundary Layer Program
- Airfoil Pressure Distribution Program
- Turbine Airfoil Design and Section Properties Program
- Airfoil Curved Line Fairing Program
- Airfoil Straight Line Fairing Program

These decks have all been converted to Fortran IV, checked-out on IBM 7094, and delivered to the Air Force along with flow diagrams and test cases. This was done under Task IIe.

(U) Engineering write-up for the first four decks listed above has been completed. Those remaining will be prepared, and a complete set delivered to the Air Force as required by this Task. The above decks, with modifications based on recent test results, are presently being utilized to design the Demonstrator Turbine for Phase IV.

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TABLE VIII  
SECOND VANE CASCADE AIRFOIL, RECAMBERING DESIGN A

Radius of FF Section = 7.5000  
Pitch = 0.5890

Axial Chord = 0.8320  
No. of Foils = 80.00

Actual Chord = 0.8634

Percent X	Foil X	Foil Y Suction	Foil Y (Circle)	Foil Y Pressure	Foil Y (Circle)
0.0	0.0	-0.2880	-0.2831	-0.2391	-0.2831
0.01	0.0083	-0.3016	-0.3016	-0.2479	-0.2682
0.02	0.0166	-0.3145	-0.3145	-0.2567	-0.2657
0.03	0.0250	-0.3269	-0.3269	-0.2655	-0.2673
0.04	0.0333	-0.3387	-0.3387	-0.2743	-0.2743
0.05	0.0416	-0.3500	-0.3500	-0.2825	-0.2825
0.10	0.0832	-0.3989	-0.3989	-0.3146	-0.3146
0.15	0.1248	-0.4368	-0.4368	-0.3353	-0.3353
0.20	0.1664	-0.4649	-0.4649	-0.3476	-0.3476
0.25	0.2080	-0.4843	-0.4843	-0.3531	-0.3531
0.30	0.2496	-0.4958	-0.4958	-0.3531	-0.3531
0.35	0.2912	-0.5001	-0.5001	-0.3482	-0.3482
0.40	0.3328	-0.4975	-0.4975	-0.3391	-0.3391
0.45	0.3744	-0.4886	-0.4886	-0.3262	-0.3262
0.50	0.4160	-0.4735	-0.4735	-0.3098	-0.3098
0.55	0.4576	-0.4526	-0.4526	-0.2901	-0.2901
0.60	0.4992	-0.4261	-0.4261	-0.2673	-0.2673
0.65	0.5408	-0.3939	-0.3939	-0.2416	-0.2416
0.70	0.5824	-0.3564	-0.3564	-0.2130	-0.2130
0.75	0.6240	-0.3134	-0.3134	-0.1817	-0.1817
0.80	0.6656	-0.2651	-0.2651	-0.1476	-0.1476
0.85	0.7072	-0.2111	-0.2111	-0.1109	-0.1109
0.90	0.7488	-0.1514	-0.1514	-0.0714	-0.0714
0.95	0.7904	-0.0854	-0.0854	-0.0291	-0.0291
0.98	0.8154	-0.0426	-0.0426	-0.0024	-0.0025
0.99	0.8237	-0.0277	-0.0277	0.0066	-0.0001
1.00	0.8320	-0.0125	-0.0100	0.0157	-0.0100
Foil L.E. Circle	X = 0.0175,	Y = -0.2831	R = 0.0175		
Foil T.E. Circle	X = 0.8220,	Y = -0.0100,	R = 0.0100		
Foil L.E. Tangency Pt. Suction	X = 0.0026,	Y = -0.2922			
Foil L.E. Tangency Pt. Pressure	X = 0.0302,	Y = -0.2711			
Foil T.E. Tangency Pt. Suction	X = 0.8308,	Y = -0.0148			
Foil T.E. Tangency Pt. Pressure	X = 0.8146,	Y = -0.0032			
Foil Nose Point	X = 0.0069,	Y = -0.2692			
Foil Tail Point	X = 0.8278,	Y = -0.0019			
Foil Area	= 0.1002				
Gaging	LAMBDA = 0.3310,	X = 0.9842,	Y = -0.3548		
Gaging Angle	= 34.190				
Center of Gravity	X = 0.4132,	Y = -0.3347			
Radial Reference	X = 0.4132,	Y = -0.3347			
Inlet Angle	= 37.330,	DELTA BETA 1 = 12.000			
Exit Angle	= 35.574,	DELTA BETA 2 = 14.000			
Uncovered Turning	= 17.212				
Min. Moment of Inertia	= 0.00046481				
Max. Moment of Inertia	= 0.00476761				
Principal Axis Angle	= -22.819				
Pitch/Axial Chord	= 0.7080				
Pitch/Actual Chord	= 0.6823				

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TABLE IX

## SECOND VANE CASCADE AIRFOIL, RECAMBERING DESIGN A

Radius of AA Section = 7.6500  
 Pitch = 0.6008

Axial Chord = 0.8357  
 No. of Foils = 80.00

Actual Chord = 0.8739

Percent X	Foil X	Foil Y Suction	Foil Y (Circle)	Foil Y Pressure	Foil Y (Circle)
0.0	-0.0083	-0.2987	-0.2937	-0.2494	-0.2937
0.01	0.0001	-0.3122	-0.3122	-0.2581	-0.2785
0.02	0.0085	-0.3251	-0.3251	-0.2667	-0.2758
0.03	0.0168	-0.3375	-0.3375	-0.2754	-0.2773
0.04	0.0252	-0.3493	-0.3493	-0.2840	-0.2840
0.05	0.0335	-0.3606	-0.3606	-0.2921	-0.2921
0.10	0.0753	-0.4101	-0.4101	-0.3250	-0.3250
0.15	0.1171	-0.4487	-0.4487	-0.3476	-0.3476
0.20	0.1589	-0.4776	-0.4776	-0.3620	-0.3620
0.25	0.2007	-0.4979	-0.4979	-0.3696	-0.3696
0.30	0.2425	-0.5102	-0.5102	-0.3712	-0.3712
0.35	0.2842	-0.5150	-0.5150	-0.3673	-0.3673
0.40	0.3260	-0.5128	-0.5128	-0.3585	-0.3585
0.45	0.3678	-0.5040	-0.5040	-0.3452	-0.3452
0.50	0.4096	-0.4888	-0.4888	-0.3277	-0.3277
0.55	0.4514	-0.4673	-0.4673	-0.3062	-0.3062
0.60	0.4932	-0.4399	-0.4399	-0.2812	-0.2812
0.65	0.5350	-0.4064	-0.4064	-0.2527	-0.2527
0.70	0.5767	-0.3671	-0.3671	-0.2211	-0.2211
0.75	0.6185	-0.3219	-0.3219	-0.1863	-0.1863
0.80	0.6603	-0.2708	-0.2708	-0.1487	-0.1487
0.85	0.7021	-0.2135	-0.2135	-0.1082	-0.1082
0.90	0.7439	-0.1497	-0.1497	-0.0650	-0.0650
0.95	0.7857	-0.0790	-0.0790	-0.0191	-0.0191
0.98	0.8107	-0.0328	-0.0328	0.0097	0.0095
0.99	0.8191	-0.0167	-0.0167	0.0194	0.0120
1.00	0.8275	-0.0003	0.0021	0.0292	0.0021
Foil L.E. Circle	X = 0.0096,	Y = -0.2937,	R = 0.0179		
Foil T.E. Circle	X = 0.8175,	Y = 0.0021,	R = 0.0100		
Foil L.E. Tangency Pt. Suction	X = -0.0056,	Y = -0.3030			
Foil L.E. Tangency Pt. Pressure	X = 0.0225,	Y = -0.2812			
Foil T.E. Tangency Pt. Suction	X = 0.8264,	Y = -0.0024			
Foil T.E. Tangency Pt. Pressure	X = 0.8099,	Y = 0.0086			
Foil Nose Point	X = -0.0013,	Y = -0.2795			
Foil Tail Point	X = 0.8230,	Y = 0.0104			
Foil Area	= 0.1008				
Gaging	LAMBDA = 0.3240,	X = 0.5788,	Y = -0.3650		
Gaging Angle	= 32.635				
Center of Gravity	X = 0.4107,	Y = -0.3436			
Radial Reference	X = 0.4132,	Y = -0.3347			
Inlet Angle	= 37.732	DELTA BETA 1 = 12.494			
Exit Angle	= 33.834	DELTA BETA 2 = 13.772			
Uncovered Turning	= 17.541				
Min. Moment of Inertia	= 0.00051391				
Max. Moment of Inertia	= 0.00501637				
Principal Axis Angle	= -24.293				
Pitch/Axial Chord	= 0.7189				
Pitch/Actual Chord	= 0.6876				

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TABLE X  
SECOND VANE CASCADE AIRFOIL, RECAMBERING DESIGN A

Radius of BB Section = 8.4863  
Pitch = 0.6665

Axial Chord = 0.8565  
No. of Foils = 80.00

Actual Chord = 0.9485

Percent X	Foil X	Foil Y Suction	Foil Y (Circle)	Foil Y Pressure	Foil Y (Circle)
0.0	-0.0543	-0.3745	-0.3683	-0.3252	-0.3683
0.01	-0.0458	-0.3869	-0.3869	-0.3324	-0.3519
0.02	-0.0372	-0.3989	-0.3989	-0.3397	-0.3485
0.03	-0.0286	-0.4105	-0.4105	-0.3469	-0.3492
0.04	-0.0201	-0.4217	-0.4217	-0.3542	-0.3542
0.05	-0.0115	-0.4325	-0.4325	-0.3613	-0.3613
0.10	0.0313	-0.4804	-0.4804	-0.3933	-0.3933
0.15	0.0741	-0.5190	-0.5190	-0.4191	-0.4191
0.20	0.1170	-0.5488	-0.5488	-0.4383	-0.4383
0.25	0.1598	-0.5701	-0.5701	-0.4504	-0.4504
0.30	0.2026	-0.5834	-0.5834	-0.4554	-0.4554
0.35	0.2454	-0.5888	-0.5888	-0.4531	-0.4531
0.40	0.2883	-0.5866	-0.5866	-0.4436	-0.4436
0.45	0.3311	-0.5767	-0.5767	-0.4270	-0.4270
0.50	0.3739	-0.5594	-0.5594	-0.4036	-0.4036
0.55	0.4167	-0.5346	-0.5346	-0.3738	-0.3738
0.60	0.4596	-0.5022	-0.5022	-0.3381	-0.3381
0.65	0.5024	-0.4621	-0.4621	-0.2970	-0.2970
0.70	0.5452	-0.4141	-0.4141	-0.2509	-0.2509
0.75	0.5880	-0.3580	-0.3580	-0.2004	-0.2004
0.80	0.6309	-0.2934	-0.2934	-0.1459	-0.1459
0.85	0.6737	-0.2197	-0.2197	-0.0878	-0.0878
0.90	0.7165	-0.1360	-0.1360	-0.0266	-0.0266
0.95	0.7593	-0.0409	-0.0409	0.0375	0.0375
0.98	0.7850	0.0226	0.0226	0.0771	0.0767
0.99	0.7936	0.0449	0.0449	0.0905	0.0796
1.00	0.8022	0.0679	0.0697	0.1038	0.0697
Foil L.E. Circle	X = -0.0343,	Y = -0.3683,	R = 0.0200		
Foil T.E. Circle	X = 0.7922,	Y = 0.0697,	R = 0.0100		
Foil L.E. Tangency Pt. Suction	X = -0.0508,	Y = -0.3796			
Foil L.E. Tangency Pt. Pressure	X = -0.0214,	Y = -0.3531			
Foil T.E. Tangency Pt. Suction	X = 0.8016,	Y = 0.0663			
Foil T.E. Tangency Pt. pressure	X = 0.7838,	Y = 0.0751			
Foil Nose Point	X = -0.0477,	Y = -0.3535			
Foil Tail Point	X = 0.7966,	Y = 0.0787			
Foil Area	= 0.1073				
Gaging Gaging Angle	LAMBDA = 0.2969, = 26.46°	X = 0.5531,	Y = -0.4044		
Center of Gravity Radial Reference	X = 0.3939, X = 0.4132,	Y = -0.3812 Y = -0.3347			
Inlet Angle	= 42.100,	DELTA BETA 1 = 15.250			
Exit Angle	= 26.450,	DELTA BETA 2 = 12.500			
Uncovered Turning	= 18.492				
Min. Moment of Inertia	= 0.00073689				
Max. Moment of Inertia	= 0.00676781				
Principal Axis Angle	= -32.600				
Pitch/Axial Chord	= 0.7782				
Pitch/Actual Chord	= 0.7027				

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TABLE XI

## SECOND VANE CASCADE AIRFOIL, RECAMBERING DESIGN A

Radius of CC Section = 9.4725  
 Pitch = 0.7440

Axial Chord = 0.8810  
 No. of Foils = 80.00

Actual Chord = 1.0682

Percent X	Foil X	Foil Y Suction	Foil Y (Circle)	Foil Y Pressure	Foil Y (Circle)
0.0	-0.1086	-0.4844	-0.4759	-0.4368	-0.4759
0.01	-0.0998	-0.4944	-0.4944	-0.4419	-0.4581
0.02	-0.0910	-0.5041	-0.5041	-0.4470	-0.4540
0.03	-0.0822	-0.5135	-0.5135	-0.4522	-0.4538
0.04	-0.0734	-0.5225	-0.5225	-0.4573	-0.4573
0.05	-0.0646	-0.5312	-0.5312	-0.4623	-0.4623
0.10	-0.0205	-0.5694	-0.5694	-0.4846	-0.4846
0.15	0.0235	-0.5994	-0.5994	-0.5019	-0.5019
0.20	0.0676	-0.6218	-0.6218	-0.5136	-0.5136
0.25	0.1116	-0.6366	-0.6366	-0.5191	-0.5191
0.30	0.1557	-0.6441	-0.6441	-0.5180	-0.5180
0.35	0.1997	-0.6445	-0.6445	-0.5099	-0.5099
0.40	0.2438	-0.6376	-0.6376	-0.4945	-0.4945
0.45	0.2878	-0.6235	-0.6235	-0.4716	-0.4716
0.50	0.3319	-0.6019	-0.6019	-0.4413	-0.4413
0.55	0.3759	-0.5727	-0.5727	-0.4037	-0.4037
0.60	0.4200	-0.5356	-0.5356	-0.3591	-0.3591
0.65	0.4640	-0.4899	-0.4899	-0.3080	-0.3080
0.70	0.5081	-0.4351	-0.4351	-0.2509	-0.2509
0.75	0.5521	-0.3703	-0.3703	-0.1882	-0.1882
0.80	0.5962	-0.2945	-0.2945	-0.1205	-0.1205
0.85	0.6402	-0.2063	-0.2063	-0.0484	-0.0484
0.90	0.6843	-0.1044	-0.1044	0.0276	0.0276
0.95	0.7283	0.0131	0.0131	0.1072	0.1072
0.98	0.7547	0.0917	0.0917	0.1564	0.1559
0.99	0.7635	0.1195	0.1195	0.1730	0.1593
1.00	0.7724	0.1479	0.1494	0.1895	0.1494
Foil L.E. Circle	X = -0.0861,	Y = -0.4759,	R = 0.0225		
Foil T.E. Circle	X = 0.7624,	Y = 0.1494,	R = 0.0100		
Foil L.E. Tangency Pt. Suction	X = -0.1031,	Y = -0.4908			
Foil L.E. Tangency Pt. Pressure	X = -0.0748,	Y = -0.4565			
Foil T.E. Tangency Pt. Suction	X = 0.7719,	Y = 0.1465			
Foil T.E. Tangency Pt. Pressure	X = 0.7535,	Y = 0.1541			
Foil Nose Point	X = -0.1035,	Y = -0.4616			
Foil Tail Point	X = 0.7662,	Y = 0.1587			
Foil Area		= 0.1178			
Gaging Angle	LAMBDA = 0.2895, = 22.900	X = 0.5185, Y = -0.3988		Y = -0.4207	
Center of Gravity Radial Reference	X = 0.3686, X = 0.4132,	Y = -0.3347			
Inlet Angle	= 50.500,	DELTA BETA 1 = 18.500			
Exit Angle	= 22.500,	DELTA BETA 2 = 11.000			
Uncovered Turning	= 18.394				
Min. Moment of Inertia	= 0.00078393				
Max. Moment of Inertia	= 0.00948326				
Principal Axis Angle	= -40.289				
Pitch/Axial Chord	= 0.8445				
Pitch/Actual Chord	= 0.6965				

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TABLE XII  
SECOND VANE CASCADE AIRFOIL, RECAMBERING DESIGN A

Radius of DD Section = 10.4588  
Pitch = 0.8214

Axial Chord = 0.9055  
No. of Foils = 80.00

Actual Chord = 1.1773

Percent	Foil X	Foil Y Suction	Foil Y (Circle)	Foil Y Pressure	Foil Y (Circle)
0.0	-0.1630	-0.5503	-0.5395	-0.5026	-0.5395
0.01	-0.1539	-0.5588	-0.5588	-0.5063	-0.5203
0.02	-0.1449	-0.5671	-0.5671	-0.5099	-0.5155
0.03	-0.1358	-0.5748	-0.5748	-0.5135	-0.5146
0.04	-0.1268	-0.5820	-0.5820	-0.5171	-0.5171
0.05	-0.1177	-0.5887	-0.5887	-0.5205	-0.5205
0.10	-0.0724	-0.6160	-0.6160	-0.5344	-0.5344
0.15	-0.0272	-0.6341	-0.6341	-0.5424	-0.5424
0.20	0.0181	-0.6443	-0.6443	-0.5442	-0.5442
0.25	0.0634	-0.6474	-0.6474	-0.5395	-0.5395
0.30	0.1087	-0.6440	-0.6440	-0.5280	-0.5280
0.35	0.1539	-0.6345	-0.6345	-0.5096	-0.5096
0.40	0.1992	-0.6188	-0.6188	-0.4843	-0.4843
0.45	0.2445	-0.5972	-0.5972	-0.4521	-0.4521
0.50	0.2898	-0.5694	-0.5694	-0.4132	-0.4132
0.55	0.3350	-0.5352	-0.5352	-0.3679	-0.3679
0.60	0.3803	-0.4940	-0.4940	-0.3164	-0.3164
0.65	0.4256	-0.4451	-0.4451	-0.2592	-0.2592
0.70	0.4709	-0.3871	-0.3871	-0.1965	-0.1965
0.75	0.5161	-0.3184	-0.3184	-0.1289	-0.1289
0.80	0.5614	-0.2367	-0.2367	-0.0567	-0.0567
0.85	0.6067	-0.1408	-0.1408	0.0197	0.0197
0.90	0.6520	-0.0305	-0.0305	0.0999	0.0999
0.95	0.6972	0.0929	0.0929	0.1835	0.1835
0.98	0.7244	0.1725	0.1725	0.2352	0.2350
0.99	0.7335	0.1998	0.1998	0.2526	0.2391
1.00	0.7425	0.2278	0.2291	0.2700	0.2291
Foil L.E. Circle	X = 0.1380	Y = -0.5395	R = 0.0250		
Foil T.E. Circle	X = 0.7325	Y = -0.2291	R = 0.0100		
Foil L.E. Tangency Pt. Suction	X = -0.1552	Y = -0.5577			
Foil L.E. Tangency Pt. Pressure	X = -0.1287	Y = -0.5163			
Foil T.E. Tangency Pt. Suction	X = 0.7420	Y = 0.2260			
Foil T.E. Tangency Pt. Pressure	X = 0.7237	Y = 0.2338			
Foil Nose Point	X = -0.1590	Y = -0.5260			
Foil Tail Point	X = 0.7364	Y = 0.2384			
Foil Area	= 0.1190				
Gaging	LAMBDA = 0.3223	X = 0.4546	Y = -0.3958		
Gaging Angle	23.100				
Center of Gravity	X = 0.3340	Y = -0.3648			
Radial Reference	X = 0.4132	Y = -0.3347			
Inlet Angle	= 57.400	DELTA BETA 1 = 21.750			
Exit Angle	= 22.750	DELTA BETA 2 = 9.500			
Uncovered Turning	= 18.360				
Min. Moment of Inertia	= 0.00062268				
Max. Moment of Inertia	= 0.01144839				
Principal Axis Angle	= -44.141				
Pitch/Axial Chord	= 0.9072				
Pitch/Actual Chord	= 0.6977				

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**TABLE XIII**  
**SECOND VANE CASCADE AIRFOIL, RECAMBERING DESIGN A**

Radius of EE Section = 10.8600 Pitch = 0.8529		Axial Chord - 0.9155 No. of Foils = 80.00		Actual Chord - 1.2091	
Percent	Foil X	Foil Y Suction	Foil Y (Circle)	Foil Y Pressure	Foil Y (Circle)
0.0	-0.1851	-0.5548	-0.5431	-0.5068	-0.5431
0.01	-0.1759	-0.5630	-0.5630	-0.5100	-0.5233
0.02	-0.1668	-0.5708	-0.5708	-0.5130	-0.5183
0.03	-0.1576	-0.5782	-0.5782	-0.5161	-0.5172
0.04	-0.1485	-0.5850	-0.5850	-0.5192	-0.5192
0.05	-0.1393	-0.5914	-0.5914	-0.5222	-0.5222
0.10	-0.0935	-0.6166	-0.6166	-0.5323	-0.5323
0.15	-0.0478	-0.6322	-0.6322	-0.5356	-0.5356
0.20	-0.0020	-0.6395	-0.6395	-0.5320	-0.5320
0.25	0.0438	-0.6393	-0.6393	-0.5217	-0.5217
0.30	0.0896	-0.6321	-0.6321	-0.5048	-0.5048
0.35	0.1353	-0.6184	-0.6184	-0.4814	-0.4814
0.40	0.1811	-0.5983	-0.5983	-0.4516	-0.4516
0.45	0.2269	-0.5719	-0.5719	-0.4158	-0.4158
0.50	0.2727	-0.5393	-0.5393	-0.3739	-0.3739
0.55	0.3184	-0.5002	-0.5002	-0.3265	-0.3265
0.60	0.3642	-0.4544	-0.4544	-0.2739	-0.2739
0.65	0.4100	-0.4012	-0.4012	-0.2162	-0.2162
0.70	0.4558	-0.3397	-0.3397	-0.1538	-0.1538
0.75	0.5015	-0.2685	-0.2685	-0.0871	-0.0871
0.80	0.5473	-0.1858	-0.1858	-0.0164	-0.0164
0.85	0.5931	-0.0908	-0.0908	0.0581	0.0581
0.90	0.6388	0.0163	0.0163	0.1360	0.1360
0.95	0.6846	0.1339	0.1339	0.2172	0.2172
0.98	0.7121	0.2086	0.2086	0.2672	0.2671
0.99	0.7212	0.2341	0.2341	0.2841	0.2715
1.00	0.7304	0.2598	0.2616	0.3010	0.2616
Foil L.E. Circle	X = -0.1591	Y = -0.5431	R = 0.0260		
Foil T.E. Circle	X = 0.7204	Y = 0.2616	R = 0.0100		
Foil L. E. Tangency Pt. Suction	X = -0.1764	Y = -0.5625			
Foil L.E. Tangency Pt. Pressure	X = -0.1507	Y = -0.5185			
Foil T.E. Tangency Pt. Suction	X = 0.7298	Y = 0.2582			
Foil T.E. Tangency Pt. Pressure	X = 0.7116	Y = 0.2663			
Foil Nose Point	X = -0.1815	Y = -0.5300			
Foil Tail Point	X = 0.7245	Y = 0.2707			
Foil Area		= 0.1220			
Gaging Gaging Angle	LAMBDA = 0.3519 = 24.368	X = 0.4306	Y = -0.3745		
Center of Gravity Radial Reference	X = 0.3059 X = 0.4132	Y = -0.3455 Y = -0.3347			
Inlet Angle	= 59.762	DELTA BETA 1 = 23.072			
Exit Angle	= 24.140	DELTA BETA 2 = 8.890			
Uncovered Turning	= 17.216				
Min. Moment of Inertia	= 0.00054959				
Max. Moment of Inertia	= 0.01220553				
Principal Axis Angle	= -44.918				
Pitch/Axial Chord	= 0.9317				
Pitch/Actual Chord	= 0.7054				

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TABLE XIV

## SECOND VANE CASCADE AIRFOIL, RECAMBERING DESIGN A

Radius of HH Section = 11.4100 Pitch = 0.8961		Axial Chord = 0.9291 No. of Foils = 80.00		Actual Chord = 1.2378	
Percent X	Foil X	Foil Y Suction	Foil Y (Circle)	Foil Y Pressure	Foil Y (Circle)
0.0	-0.2154	-0.5379	-0.5251	-0.4895	-0.5251
0.01	-0.2061	-0.5457	-0.5457	-0.4920	-0.5045
0.02	-0.1968	-0.5532	-0.5532	-0.4944	-0.4991
0.03	-0.1875	-0.5603	-0.5603	-0.4968	-0.4977
0.04	-0.1782	-0.5670	-0.5670	-0.4993	-0.4993
0.05	-0.1689	-0.5733	-0.5733	-0.5014	-0.5014
0.10	-0.1225	-0.5987	-0.5987	-0.5056	-0.5056
0.15	-0.0760	-0.6145	-0.6145	-0.5007	-0.5007
0.20	-0.0295	-0.6210	-0.6210	-0.4879	-0.4879
0.25	0.0169	-0.6187	-0.6187	-0.4683	-0.4683
0.30	0.0634	-0.6078	-0.6078	-0.4428	-0.4428
0.35	0.1098	-0.5887	-0.5887	-0.4117	-0.4117
0.40	0.1563	-0.5616	-0.5616	-0.3757	-0.3757
0.45	0.2027	-0.5269	-0.5269	-0.3352	-0.3352
0.50	0.2492	-0.4847	-0.4847	-0.2903	-0.2903
0.55	0.2956	-0.4353	-0.4353	-0.2416	-0.2416
0.60	0.3421	-0.3790	-0.3790	-0.1892	-0.1892
0.65	0.3886	-0.3158	-0.3158	-0.1334	-0.1334
0.70	0.4350	-0.2460	-0.2460	-0.0742	-0.0742
0.75	0.4815	-0.1696	-0.1696	-0.0119	-0.0119
0.80	0.5279	-0.0868	-0.0868	0.0533	0.0533
0.85	0.5744	0.0023	0.0023	0.1214	0.1214
0.90	0.6208	0.0974	0.0974	0.1923	0.1923
0.95	0.6673	0.1981	0.1981	0.2658	0.2658
0.98	0.6952	0.2610	0.2610	0.3112	0.3112
0.99	0.7045	0.2824	0.2824	0.3264	0.3160
1.00	0.7138	0.3039	0.3060	0.3417	0.3060
Foil L.E. Circle	X = -0.1880	Y = -0.5251	R = 0.0274		
Foil T.E. Circle	X = 0.7038	Y = 0.3060	R = 0.0100		
Foil L.E. Tangency Pt. Suction	X = -0.2055	Y = -0.5462			
Foil L. E. Tangency Pt. Pressure	X = -0.1809	Y = -0.4986			
Foil T. E. Tangency Pt. Suction	X = 0.7125	Y = -0.3021			
Foil T.E. Tangency Pt. Pressure	X = 0.6952	Y = 0.3112			
Foil Nose Point	X = -0.2123	Y = -0.5125			
Foil Nose Point	X = 0.7083	Y = 0.3149			
Foil Area	= 0.1311				
Gaging	LAMBDA = 0.4060	X = 0.3681	Y = -0.3444		
Gaging Angle	= 26.937				
Center of Gravity	X = 0.2515	Y = -0.3179			
Radial Reference	X = 0.4432	Y = -0.3347			
Inlet Angle	= 62.722	DELTA BETA 1 = 24.885			
Exit Angle	= 27.322	DELTA BETA 2 = 8.053			
Uncovered Turning	= 12.843				
Min. Moment of Inertia	= 0.00047562				
Max. Moment of Inertia	= 0.01310568				
Principal Axis Angle	= -45.641				
Pitch/Axial Chord	= 0.9645				
Pitch/Actual Chord	= 0.7240				

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TABLE XV  
SECOND VANE CASCADE AIRFOIL, RECAMBERING DESIGN A

Radius of GG Section - 11.4450 Pitch = 0.8989		Axial Chord = 0.9300 No. of Foils = 80.00		Actual Chord = 1.2390	
Percent X	Foil X	Foil Y Suction	Foil Y (Circle)	Foil Y Pressure	Foil Y (Circle)
0.0	-0.2173	-0.5359	-0.5230	-0.4874	-0.5230
0.01	-0.2080	-0.5436	-0.5436	-0.4898	-0.5024
0.02	-0.1987	-0.5512	-0.5512	-0.4922	-0.4970
0.03	-0.1894	-0.5583	-0.5583	-0.4947	-0.4955
0.04	-0.1801	-0.5650	-0.5650	-0.4971	-0.4971
0.05	-0.1708	-0.5713	-0.5713	-0.4991	-0.4991
0.10	-0.1243	-0.5969	-0.5969	-0.5029	-0.5029
0.15	-0.0778	-0.6128	-0.6128	-0.4974	-0.4974
0.20	-0.0313	-0.6194	-0.6194	-0.4840	-0.4840
0.25	0.0152	-0.6170	-0.6170	-0.4638	-0.4638
0.30	0.0617	-0.6059	-0.6059	-0.4376	-0.4376
0.35	0.1082	-0.5865	-0.5865	-0.4060	-0.4060
0.40	0.1547	-0.5589	-0.5589	-0.3696	-0.3696
0.45	0.2012	-0.5236	-0.5236	-0.3288	-0.3288
0.50	0.2477	-0.4806	-0.4806	-0.2838	-0.2838
0.55	0.2942	-0.4305	-0.4305	-0.2351	-0.2351
0.60	0.3407	-0.3733	-0.3733	-0.1828	-0.1828
0.65	0.3872	-0.3094	-0.3094	-0.1271	-0.1271
0.70	0.4337	-0.2389	-0.2389	-0.0683	-0.0683
0.75	0.4802	-0.1621	-0.1621	-0.0064	-0.0064
0.80	0.5267	-0.0793	-0.0793	0.0583	0.0583
0.85	0.5732	0.0092	0.0092	0.1259	0.1259
0.90	0.6197	0.1032	0.1032	0.1961	0.1961
0.95	0.6662	0.2025	0.2025	0.2690	0.2690
0.98	0.6941	0.2645	0.2645	0.3139	0.3139
0.99	0.7034	0.2855	0.2855	0.3291	0.3188
1.00	0.7127	0.3067	0.3088	0.3442	0.3088
Foil L.E. Circle	X = -0.1898	Y = -0.5230	R = 0.0275		
Foil T.E. Circle	X = 0.7027	Y = 0.3088	R = 0.0100		
Foil L.E. Tangency Pt. Suction	X = -0.2073	Y = -0.5442			
Foil L.E. Tangency Pt. Pressure	X = -0.1829	Y = -0.4964			
Foil T.E. Tangency Pt. Suction	X = 0.7119	Y = 0.3048			
Foil T.E. Tangency Pt. Pressure	X = 0.6942	Y = 0.3141			
Foil Nose Point	X = -0.2143	Y = -0.5104			
Foil Tail Point	X = 0.7073	Y = 0.3177			
Foil Area	= 0.1319				
Gaging Angle	LAMBDA = 0.4098 = 27.119	X = 0.3633	Y = -0.3431		
Center of Gravity Radial Reference	X = 0.2475 X = 0.4132	Y = -0.3161 Y = -0.3347			
Inlet Angle	= 62.900	DELTA BETA 1 = 25.000			
Exit Angle	= 27.574	DELTA BETA 2 = 8.000			
Uncovered Turning	= 12.470				
Min. Moment of Inertia	= 0.00047307				
Max. Moment of Inertia	= 0.01314440				
Principal Axis Angle	= -45.681				
Pitch/Axial Chord	= 0.9665				
Pitch/Actual Chord	= 0.7255				

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TABLE XVI

## SECOND VANE CASCADE AIRFOIL, RECAMBERING DESIGN B

Radius of FF Section = 7.5000 Pitch = 0.5890		Axial Chord = 0.8320 No. of Foils = 80.00		Actual Chord = 0.8485	
Percent X	Foil X	Foil Y Suction	Foil Y (Circle)	Foil Y Pressure	Foil Y (Circle)
0.0	0.0	-0.2402	-0.2346	-0.1842	-0.2346
0.01	0.0083	-0.2537	-0.2537	-0.1930	-0.2183
0.02	0.0166	-0.2665	-0.2665	-0.2018	-0.2148
0.03	0.0250	-0.2788	-0.2788	-0.2107	-0.2152
0.04	0.0333	-0.2904	-0.2904	-0.2195	-0.2196
0.05	0.0416	-0.3014	-0.3014	-0.2281	-0.2281
0.10	0.0832	-0.3489	-0.3489	-0.2640	-0.2640
0.15	0.1248	-0.3855	-0.3855	-0.2898	-0.2898
0.20	0.1664	-0.4129	-0.4129	-0.3076	-0.3076
0.25	0.2080	-0.4325	-0.4325	-0.3187	-0.3187
0.30	0.2496	-0.4449	-0.4449	-0.3241	-0.3241
0.35	0.2912	-0.4507	-0.4507	-0.3243	-0.3243
0.40	0.3328	-0.4504	-0.4504	-0.3200	-0.3200
0.45	0.3744	-0.4440	-0.4440	-0.3114	-0.3114
0.50	0.4160	-0.4318	-0.4318	-0.2988	-0.2988
0.55	0.4576	-0.4138	-0.4138	-0.2825	-0.2825
0.60	0.4992	-0.3898	-0.3898	-0.2627	-0.2627
0.65	0.5408	-0.3597	-0.3597	-0.2394	-0.2394
0.70	0.5824	-0.3231	-0.3231	-0.2128	-0.2128
0.75	0.6240	-0.2798	-0.2798	-0.1828	-0.1828
0.80	0.6656	-0.2310	-0.2310	-0.1496	-0.1496
0.85	0.7072	-0.1788	-0.1788	-0.1131	-0.1131
0.90	0.7488	-0.1246	-0.1246	-0.0733	-0.0733
0.95	0.7904	-0.0693	-0.0693	-0.0300	-0.0300
0.98	0.8154	-0.0357	-0.0357	-0.0024	-0.0025
0.99	0.8237	-0.0245	-0.0245	0.0070	-0.0001
1.00	0.8320	-0.0133	-0.0100	0.0164	-0.0100
Foil L. E. Circle	X = 0.0200	Y = -0.2346	R = 0.0200		
Foil T. E. Circle	X = 0.8220	Y = -0.0100	R = 0.0100		
Foil L. E. Tangency Pt. Suction	X = 0.0029	Y = -0.2450			
Foil L. E. Tangency Pt. Pressure	X = 0.0345	Y = -0.2208			
Foil T. E. Tangency Pt. Suction	X = 0.8300	Y = -0.0159			
Foil T. E. Tangency Pt. Pressure	X = 0.8145	Y = -0.0034			
Foil Nose Point	X = 0.0079	Y = -0.2187			
Foil Tail Point	X = 0.8283	Y = -0.0022			
Foil Area	0.0817				
Gaging Gaging Angle	LAMBDA = 0.3552 = 37.082	X = 0.5738	Y = -0.3312		
Center of Gravity Radial Reference	X = 0.3924 X = 0.3924	Y = -0.3113 Y = -0.3113			
Inlet Angle	= 37.330	DELTA BETA 1 = 12.000			
Exit Angle	= 39.000	DELTA BETA 2 = 5.000			
Uncovered Turning	= 10.654				
Min. Moment of Inertia	= 0.00038949				
Max. Moment of Inertia	= 0.00367671				
Principal Axis Angle	= -17.113				
Pitch/Axial Chord	= 0.7080				
Pitch/Actual Chord	= 0.6942				

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TABLE XVII

## SECOND VANE CASCADE AIRFOIL, RECAMBERING DESIGN B

Radius of AA Section = 7.6500 Pitch = 0.6008		Axial Chord=0.8357 No. of Foils=80.00		Actual Chord 0.8609	
Percent X	Foil X	Foil Y Section	Foil Y (Circle)	Foil Y Pressure	Foil Y (Circle)
0.0	-0.0083	-0.2619	-0.2562	-0.2057	-0.2562
0.01	0.0001	-0.2754	-0.2754	-0.2144	-0.2397
0.02	0.0085	-0.2882	-0.2882	-0.2231	-0.2362
0.03	0.0168	-0.3003	-0.3003	-0.2317	-0.2364
0.04	0.0252	-0.3119	-0.3119	-0.2404	-0.2405
0.05	0.0335	-0.3228	-0.3228	-0.2488	-0.2488
0.10	0.0753	-0.3700	-0.3700	-0.2840	-0.2840
0.15	0.1171	-0.4063	-0.4063	-0.3090	-0.3090
0.20	0.1589	-0.4334	-0.4334	-0.3257	-0.3257
0.25	0.2007	-0.4526	-0.4526	-0.3355	-0.3355
0.30	0.2425	-0.4646	-0.4646	-0.3394	-0.3394
0.35	0.2842	-0.4702	-0.4702	-0.3381	-0.3381
0.40	0.3260	-0.4696	-0.4696	-0.3321	-0.3321
0.45	0.3678	-0.4630	-0.4630	-0.3219	-0.3219
0.50	0.4096	-0.4507	-0.4507	-0.3078	-0.3078
0.55	0.4514	-0.4325	-0.4325	-0.2900	-0.2900
0.60	0.4932	-0.4083	-0.4083	-0.2687	-0.2687
0.65	0.5350	-0.3777	-0.3777	-0.2440	-0.2440
0.70	0.5767	-0.3402	-0.3402	-0.2160	-0.2160
0.75	0.6185	-0.2951	-0.2951	-0.1844	-0.1844
0.80	0.6603	-0.2430	-0.2430	-0.1494	-0.1494
0.85	0.7021	-0.1859	-0.1859	-0.1105	-0.1105
0.90	0.7439	-0.1256	-0.1256	-0.0678	-0.0678
0.95	0.7857	-0.0636	-0.0636	-0.0207	-0.0207
0.98	0.8107	-0.0260	-0.0260	0.0098	0.0095
0.99	0.8191	-0.0135	-0.0135	0.0202	0.0120
1.00	0.8275	-0.0009	0.0021	0.0306	0.0021
Foil L.E. Circle	X=0.0121,	Y = -0.2562	R=0.0204		
Foil T.E. Circle	X=0.8175	Y = 0.0021	R=0.0100		
Foil L. E. Tangency Pt. Suction	X = -0.0052	Y = -0.2669			
Foil L. E. Tangency Pt. Pressure	X = 0.0268	Y = -0.2421			
Foil T. E. Tangency Pt. Suction	X = 0.8258	Y = -0.0035			
Foil T. E. Tangency Pt. Pressure	X = 0.8097	Y = 0.0084			
Foil Nose Point	X = -0.0004	Y = -0.2401			
Foil Tail Point	X = 0.8234	Y = 0.0102			
Foil Area	=0.0880				
Gaging Gaging Angle	LAMBDA =0.3432 =34.829	X=0.5720	Y = -0.3448		
Center of Gravity Radial Reference	X=0.3938 X=0.3924	Y = -0.3225 Y = -0.3113			
Inlet Angle Exit Angle Uncovered Turning	=37.732 =36.513 =12.020	DELTA BETA 1 = 12.361 DELTA BETA 2 = 5.120			
Min. Moment of Inertia Max. Moment of Inertia	=0.00043744 =0.00408222				
Principal Axis Angle	= -19.374				
Pitch/Axial Chord Pitch/Actual Chord	=0.7189 =0.6979				

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TABLE XVIII

## SECOND VANE CASCADE AIRFOIL, RECAMBERING DESIGN B

Radius of BB Section = 8.4863 Pitch = 0.6665		Axial Chord = 0.8565 No. of Foils = 80.00		Actual Chord = 0.9463	
Percent X	Foil X	Foil Y Section	Foil Y (Circle)	Foil Y Pressure	Foil Y (Circle)
0.0	-0.0543	-0.3741	-0.3670	-0.3180	-0.3670
0.01	-0.0458	-0.3863	-0.3863	-0.3254	-0.3493
0.02	-0.0372	-0.3981	-0.3981	-0.3327	-0.3452
0.03	-0.0286	-0.4093	-0.4093	-0.3401	-0.3447
0.04	-0.0201	-0.4201	-0.4201	-0.3474	-0.3478
0.05	-0.0115	-0.4304	-0.4304	-0.3547	-0.3547
0.10	0.0313	-0.4756	-0.4756	-0.3853	-0.3853
0.15	0.0741	-0.6113	-0.5113	-0.4073	-0.4073
0.20	0.1170	-0.5385	-0.5385	-0.4216	-0.4216
0.25	0.1598	-0.5581	-0.5581	-0.4292	-0.4292
0.30	0.2026	-0.5704	-0.5704	-0.4305	-0.4305
0.35	0.2454	-0.5759	-0.5759	-0.4261	-0.4261
0.40	0.2883	-0.5747	-0.5747	-0.4162	-0.4162
0.45	0.3311	-0.5668	-0.5668	-0.4011	-0.4011
0.50	0.3739	-0.5520	-0.5520	-0.3809	-0.3809
0.55	0.4167	-0.5302	-0.5302	-0.3557	-0.3557
0.60	0.4596	-0.5008	-0.5008	-0.3256	-0.3256
0.65	0.5024	-0.4633	-0.4633	-0.2905	-0.2905
0.70	0.5452	-0.4166	-0.4166	-0.2504	-0.2504
0.75	0.5880	-0.3593	-0.3593	-0.2052	-0.2052
0.80	0.6309	-0.2904	-0.2904	-0.1546	-0.1546
0.85	0.6737	-0.2109	-0.2109	-0.0983	-0.0983
0.90	0.7165	-0.1231	-0.1231	-0.0360	-0.0360
0.95	0.7593	-0.0297	-0.0297	0.0328	0.0328
0.98	0.7850	0.0283	0.0283	0.0774	0.0767
0.99	0.7936	0.0479	0.0479	0.0927	0.0927
1.00	0.8022	0.0676	0.0697	0.1080	0.0697
Foil L. E. Circle	X = -0.0318	Y = -0.3670	R = 0.0225		
Foil T. E. Circle	X = 0.7922	Y = 0.0697	R = 0.0100		
Foil L. E. Tangency Pt. Suction	X = -0.0503	Y = -0.3799			
Foil L. E. Tangency Pt. Pressure	X = -0.0172	Y = -0.3499			
Foil T. E. Tangency Pt. Suction	X = 0.8013	Y = 0.0657			
Foil T. E. Tangency Pt. Pressure	X = 0.7834	Y = 0.0746			
Foil Nose Point	X = -0.0469	Y = -0.3503			
Foil Tail Point	X = 0.7966	Y = 0.0787			
Foil Area	= 0.1100				
Gaging Gaging Angle	LAMBDA = 0.2953 = 26.301	X = 0.5532	Y = -0.4068		
Center of Gravity Radial Reference	X = 0.3802 X = 0.3924	Y = -0.3847 Y = -0.3113			
Inlet Angle	= 42.100	DELTA BETA 1 = 14.500			
Exit Angle	= 26.350	DELTA BETA 2 = 5.800			
Uncovered Turning	= 15.099				
Min. Moment of Inertia	= 0.00072721				
Max. Moment of Inertia	= 0.00627873				
Principal Axis Angle	= -30.144				
Pitch/Axial Chord	= 0.7782				
Pitch/Actual Chord	= 0.7043				

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TABLE XIX

## SECOND VANE CASCADE AIRFOIL, RECAMBERING DESIGN B

Radius of CC Section = 9.4725  
 Pitch = 0.7440

Axial Chord = 0.8810  
 No. of Foils = 80.00

Actual Chord = 1.0657

Percent X	Foil X	Foil Y Suction	Foil Y (Circle)	Foil Y Pressure	Foil Y (Circle)
0.0	-0.1086	-0.4837	-0.4741	-0.4300	-0.4741
0.01	-0.0998	-0.4935	-0.4935	-0.4353	-0.4550
0.02	-0.0910	-0.5031	-0.5031	-0.4405	-0.4502
0.03	-0.0822	-0.5126	-0.5126	-0.4458	-0.4491
0.04	-0.0734	-0.5218	-0.5218	-0.4511	-0.4513
0.05	-0.0646	-0.5309	-0.5309	-0.4563	-0.4563
0.10	-0.0205	-0.5726	-0.5726	-0.4806	-0.4806
0.15	0.0235	-0.6084	-0.6084	-0.5012	-0.5012
0.20	0.0676	-0.6376	-0.6376	-0.5175	-0.5175
0.25	0.1116	-0.6599	-0.6599	-0.5290	-0.5290
0.30	0.1557	-0.6746	-0.6746	-0.5348	-0.5348
0.35	0.1997	-0.6812	-0.6812	-0.5342	-0.5342
0.40	0.2438	-0.6793	-0.6793	-0.5263	-0.5263
0.45	0.2878	-0.6682	-0.6682	-0.5106	-0.5106
0.50	0.3319	-0.6475	-0.6475	-0.4863	-0.4863
0.55	0.3759	-0.6168	-0.6168	-0.4532	-0.4532
0.60	0.4200	-0.5757	-0.5757	-0.4110	-0.4110
0.65	0.4640	-0.5238	-0.5238	-0.3600	-0.3600
0.70	0.5081	-0.4611	-0.4611	-0.3005	-0.3005
0.75	0.5521	-0.3872	-0.3872	-0.2332	-0.2332
0.80	0.5962	-0.3024	-0.3024	-0.1587	-0.1587
0.85	0.6402	-0.2065	-0.2065	-0.0778	-0.0778
0.90	0.6843	-0.0996	-0.0996	0.0087	0.0087
0.95	0.7283	0.0186	0.0186	0.1000	0.1000
0.98	0.7547	0.0948	0.0948	0.1568	0.1559
0.99	0.7635	0.1211	0.1211	0.1759	0.1593
1.00	0.7724	0.1478	0.1494	0.1951	0.1494
Foil L. E. Circle	X = -0.0836	Y = -0.4741	R = 0.0250		
Foil T. E. Circle	X = 0.7624	Y = 0.1494	R = 0.0100		
Foil L. E. Tangency Pt. Suction	X = -0.1023	Y = -0.4908			
Foil L. E. Tangency Pt. Pressure	X = -0.0708	Y = -0.4526			
Foil T. E. Tangency Pt. Suction	X = 0.7719	Y = 0.1463			
Foil T. E. Tangency Pt. Pressure	X = 0.7533	Y = 0.1536			
Foil Nose Point	X = -0.1029	Y = -0.4582			
Foil Tail Point	X = 0.7660	Y = 0.1587			
Foil Area	= 0.1138				
Gaging Angle	LAMBDA = 0.2763 = 21.8000	X = 0.5189	Y = -0.4440		
Center of Gravity Radial Reference	X = 0.3454 X = 0.3924	Y = -0.4429 Y = -0.3113			
Inlet Angle	= 50.500	DELTA BETA 1 = 17.250			
Exit Angle	= 21.400	DELTA BETA 2 = 6.500			
Uncovered Turning	= 13.718				
Min. Moment of Inertia	= 0.00099224				
Max. Moment of Inertia	= 0.00893008				
Principal Axis Angle	= -39.911				
Pitch/Axial Chord	= 0.8445				
Pitch/Actual Chord	= 0.6981				

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TABLE XX

SECOND VANE CASCADE AIRFOIL, RECAMBERING DESIGN B

Radius of DD Section = 10.4588 Pitch = 0.8214		Axial Chord = 0.9055 No. of Foils = 80.00	Actual Chord = 1.1746		
Percent X	Foil X	Foil Y Suction	Foil Y (Circle)	Foil Y Pressure	Foil Y (Circle)
0.0	-0.1630	-0.5494	-0.5373	-0.4961	-0.5373
0.01	-0.1539	-0.5577	-0.5577	-0.4998	-0.5169
0.02	-0.1449	-0.5659	-0.5659	-0.5036	-0.5115
0.03	-0.1358	-0.5739	-0.5739	-0.5074	-0.5098
0.04	-0.1268	-0.5817	-0.5817	-0.5112	-0.5112
0.05	-0.1177	-0.5892	-0.5892	-0.5149	-0.5149
0.10	-0.0724	-0.6234	-0.6234	-0.5317	-0.5317
0.15	-0.0272	-0.6514	-0.6514	-0.5449	-0.5449
0.20	0.0181	-0.6726	-0.6726	-0.5538	-0.5538
0.25	0.0634	-0.6866	-0.6866	-0.5576	-0.5576
0.30	0.1087	-0.6928	-0.6928	-0.5555	-0.5555
0.35	0.1539	-0.6907	-0.6907	-0.5465	-0.5465
0.40	0.1992	-0.6799	-0.6799	-0.5296	-0.5296
0.45	0.2445	-0.6599	-0.6599	-0.5042	-0.5042
0.50	0.2898	-0.6303	-0.6303	-0.4698	-0.4698
0.55	0.3350	-0.5908	-0.5908	-0.4262	-0.4262
0.60	0.3803	-0.5411	-0.5411	-0.3737	-0.3737
0.65	0.4256	-0.4813	-0.4813	-0.3129	-0.3129
0.70	0.4709	-0.4111	-0.4111	-0.2446	-0.2446
0.75	0.5161	-0.3308	-0.3308	-0.1697	-0.1697
0.80	0.5614	-0.2403	-0.2403	-0.0891	-0.0891
0.85	0.6067	-0.1395	-0.1395	-0.0038	-0.0038
0.90	0.6520	-0.0282	-0.0282	0.0856	0.0856
0.95	0.6972	0.0940	0.0940	0.1783	0.1783
0.98	0.7244	0.1727	0.1727	0.2354	0.2350
0.99	0.7335	0.1999	0.1999	0.2545	0.2391
1.00	0.7425	0.2276	0.2276	0.2291	0.2291
Foil L. E. Circle	X = -0.1355	Y = -0.5373	R = 0.0275		
Foil T. E. Circle	X = 0.7325	Y = 0.2291	R = 0.0100		
Foil L. E. Tangency Pt. Suction	X = -0.1541	Y = -0.5575			
Foil L. E. Tangency Pt. Pressure	X = -0.1249	Y = -0.5119			
Foil T. E. Tangency Pt. Suction	X = 0.7420	Y = 0.2260			
Foil T. E. Tangency Pt. Pressure	X = 0.7235	Y = 0.2334			
Foil Nose Point	X = -0.1586	Y = -0.5225			
Foil Tail Point	X = 0.7362	Y = 0.2384			
Foil Area	0.1183				
Gaging Gaging Angle	LAMBDA = 0.3077 = 22.000	X = 0.4621	Y = -0.4225		
Center of Gravity Radial Reference	X = 0.3085 X = 0.3924	Y = -0.4201 Y = -0.3113			
Inlet Angle	= 57.400	DELTA BETA 1 = 20.000			
Exit Angle	= 21.650	DELTA BETA 2 = 7.300			
Uncovered Turning	13.706				
Min. Moment of Inertia	= 0.00085047				
Max. Moment of Inertia	= 0.01156296				
Principal Axis Angle	= -44.723				
Pitch/Axial Chord	= 0.9072				
Pitch/Actual Chord	= 0.6993				

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TABLE XXI  
SECOND VANE CASCADE AIRFOIL, RECAMBERING DESIGN B

Radius of EE Section = 10.8600 Pitch = 0.8529		Axial Chord = 0.9155 No. of Foils = 80.00		Actual Chord = 1.2058		
Percent X	Foil X	Foil Y Suction	Foil Y (Circle)	Foil Y Pressure	Foil Y (Circle)	
0.0	-0.1851	-0.5527	-0.5396	-0.4991	-0.5396	
0.01	-0.1759	-0.5606	-0.5606	-0.5024	-0.5187	
0.02	-0.1668	-0.5683	-0.5683	-0.5057	-0.5130	
0.03	-0.1576	-0.5757	-0.5757	-0.5089	-0.5111	
0.04	-0.1485	-0.5827	-0.5827	-0.5122	-0.5123	
0.05	-0.1393	-0.5895	-0.5895	-0.5154	-0.5154	
0.10	-0.0935	-0.6179	-0.6179	-0.5271	-0.5271	
0.15	-0.0478	-0.6382	-0.6382	-0.5326	-0.5326	
0.20	-0.0020	-0.6508	-0.6508	-0.5325	-0.5325	
0.25	0.0438	-0.6557	-0.6557	-0.5268	-0.5268	
0.30	0.0896	-0.6527	0.6527	-0.5150	-0.5150	
0.35	0.1353	-0.6419	-0.6419	-0.4971	-0.4971	
0.40	0.1811	-0.6231	-0.6231	-0.4721	-0.4721	
0.45	0.2269	-0.5960	-0.5960	-0.4402	-0.4402	
0.50	0.2727	-0.5607	-0.5607	-0.4008	-0.4008	
0.55	0.3184	-0.5169	-0.5169	-0.3540	-0.3540	
0.60	0.3642	-0.4646	-0.4646	-0.3003	-0.3003	
0.65	0.4100	-0.4037	-0.4037	-0.2402	-0.2402	
0.70	0.4558	-0.3343	-0.3343	-0.1744	-0.1744	
0.75	0.5015	-0.2566	-0.2566	-0.1037	-0.1037	
0.80	0.5473	-0.1703	-0.1703	-0.0287	-0.0287	
0.85	0.5931	-0.0757	-0.0757	0.0498	0.0498	
0.90	0.6388	0.0274	0.0274	0.1315	0.1315	
0.95	0.6846	0.1391	0.1391	0.2157	0.2157	
0.98	0.7121	0.2104	0.2104	0.2673	0.2671	
0.99	0.7212	0.2348	0.2348	0.2847	0.2715	
1.00	0.7304	0.2597	0.2616	0.3020	0.2616	
Foil L.E. Circle	X = -0.1566	Y = -0.5396	R = 0.0285			
Foil T.E. Circle	X = 0.7204	Y = 0.2616	R = 0.0100			
Foil L.E. Tangency Pt. Suction	X = -0.1752	Y = -0.5612				
Foil L.E. Tangency Pt. Pressure	X = -0.1469	Y = -0.5128				
Foil T.E. Tangency Pt. Suction	X = 0.7297	Y = 0.2580				
Foil T.E. Tangency Pt. Pressure	X = 0.7116	Y = 0.2663				
Foil Nose Point	X = -0.1812	Y = -0.5253				
Foil Tail Point	X = 0.7246	Y = 0.2707				
Foil Area	= 0.1167					
Gaging	LAMBDA = 0.3521	X = 0.4210	Y = -0.3877			
Gaging Angle	= 24.380					
Center of Gravity	X = 0.2854	Y = -0.3740				
Radial Reference	X = 0.3924	Y = -0.3113				
Inlet Angle	= 59.762	DELTA BETA 1 = 21.053				
Exit Angle	= 24.603	DELTA BETA 2 = 7.581				
Uncovered Turning	= 13.464					
Min. Moment of Inertia	= 0.00061387					
Max. Moment of Inertia	= 0.01185749					
Principal Axis Angle	= -45.026					
Pitch/Axial Chord	= 0.9317					
Pitch/Actual Chord	= 0.7074					

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TABLE XXII

## SECOND VANE CASCADE AIRFOIL, RECAMBERING DESIGN B

Radius of HH Section = 11.4100 Pitch = 0.8961		Axial Chord = 0.9291 No. of Foils = 80.00		Actual Chord = 1.2317		
Percent X	Foil X	Foil Y Suction	Foil Y (Circle)	Foil Y Pressure	Foil Y (Circle)	
0.0	-0.2154	-0.5314	-0.5169	-0.4772	-0.5169	
0.01	-0.2061	-0.5387	-0.5386	-0.4798	-0.4953	
0.02	-0.1968	-0.5459	-0.5459	-0.4825	-0.4892	
0.03	-0.1875	-0.5524	-0.5524	-0.4852	-0.4871	
0.04	-0.1782	-0.5581	-0.5581	-0.4879	-0.4879	
0.05	-0.1689	-0.5633	-0.5633	-0.4902	-0.4902	
0.10	-0.1225	-0.5807	-0.5807	-0.4922	-0.4922	
0.15	-0.0760	-0.5857	-0.5857	-0.4820	-0.4820	
0.20	-0.0295	-0.5806	-0.5806	-0.4629	-0.4629	
0.25	0.0169	-0.5670	-0.5670	-0.4371	-0.4371	
0.30	0.0634	-0.5455	-0.5455	-0.4058	-0.4058	
0.35	0.1098	-0.5173	-0.5173	-0.3700	-0.3700	
0.40	0.1563	-0.4827	-0.4827	-0.3301	-0.3301	
0.45	0.2027	-0.4423	-0.4423	-0.2869	-0.2869	
0.50	0.2492	-0.3967	-0.3967	-0.2406	-0.2406	
0.55	0.2956	-0.3459	-0.3459	-0.1914	-0.1914	
0.60	0.3421	-0.2903	-0.2903	-0.1398	-0.1398	
0.65	0.3886	-0.2300	-0.2300	-0.0860	-0.0860	
0.70	0.4350	-0.1654	-0.1654	-0.0301	-0.0301	
0.75	0.4815	-0.0966	-0.0966	0.0275	0.0275	
0.80	0.5279	-0.0237	-0.0237	0.0868	0.0868	
0.85	0.5744	0.0528	0.0528	0.1474	0.1474	
0.90	0.6208	0.1330	0.1330	0.2095	0.2095	
0.95	0.6673	0.2166	0.2166	0.2727	0.2727	
0.98	0.6952	0.2684	0.2684	0.3112	0.3112	
0.99	0.7045	0.2859	0.2859	0.3241	0.3160	
1.00	0.7138	0.3035	0.3060	0.3370	0.3060	
Foil L.E. Circle	X = -0.1855,	Y = -0.5169,	R = 0.0299			
Foil T.E. Circle	X = 0.7038	Y = 0.3060,	R = 0.0100			
Foil L.E. Tangency PT. Suction	X = -0.2041	Y = -0.5403				
Foil L.E. Tangency PT. Pressure	X = -0.1772	Y = -0.4882				
Foil T.E. Tangency PT. Suction	X = 0.7126	Y = 0.3013				
Foil T.E. Tangency PT. Pressure	X = 0.6957	Y = 0.3119				
Foil Nose Point	X = -0.2120	Y = -0.5032				
Foil Tail Point	X = 0.7091	Y = 0.3145				
Foil Area	= 0.1087					
Gaging Angle	LAMBDA = 0.4598 = 30.866	X = 0.3379	Y = -0.2955			
Center of Gravity	X = 0.2041	Y = -0.2799				
Radial Reference	X = 0.3924	Y = -0.3113				
Inlet Angle	= 62.722	DELTA BETA 1 = 22.416				
Exit Angle	= 31.966	DELTA BETA 2 = 7.975				
Uncovered Turning	= 10.931					
Min. Moment of Inertia	= 0.00025003					
Max. Moment of Inertia	= 0.01117726					
Principal Axis Angle	= -44.941					
Pitch/Axial Chord	= 0.9648					
Pitch/Actual Chord	= 0.7276					

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TABLE XXIII

## SECOND VANE CASCADE AIRFOIL, RECAMBERING DESIGN B

Radius of CG Section = 11.4450 Pitch = 0.8989		Axial Chord = 0.9300 No. of Foils = 80.00		Actual Chord = 1.2326		
Percent X	Foil X	Foil Y Suction	Foil Y (Circle)	Foil Y Pressure	Foil Y (Circle)	
0.0	-0.2173	-0.5289	-0.5144	-0.4747	-0.5144	
0.01	-0.2080	-0.5363	-0.5361	-0.4774	-0.4927	
0.02	-0.1987	-0.5434	-0.5434	-0.4800	-0.4867	
0.03	-0.1894	-0.5498	-0.5498	-0.4826	-0.4845	
0.04	-0.1801	-0.5555	-0.5555	-0.4853	-0.4853	
0.05	-0.1708	-0.5606	-0.5606	-0.4876	-0.4876	
0.10	-0.1243	-0.5770	-0.5770	-0.4889	-0.4889	
0.15	-0.0778	-0.5810	-0.5810	-0.4774	-0.4774	
0.20	-0.0313	-0.5746	-0.5746	-0.4569	-0.4569	
0.25	0.0152	-0.5594	-0.5594	-0.4295	-0.4295	
0.30	0.0617	-0.5366	-0.5366	-0.3967	-0.3967	
0.35	0.1082	-0.5070	-0.5070	-0.3595	-0.3595	
0.40	0.1547	-0.4713	-0.4713	-0.3186	-0.3186	
0.45	0.2012	-0.4299	-0.4299	-0.2745	-0.2745	
0.50	0.2477	-0.3835	-0.3835	-0.2277	-0.2277	
0.55	0.2942	-0.3322	-0.3322	-0.1785	-0.1785	
0.60	0.3407	-0.2763	-0.2763	-0.1272	-0.1272	
0.65	0.3872	-0.2162	-0.2162	-0.0740	-0.0740	
0.70	0.4337	-0.1521	-0.1521	-0.0190	-0.0190	
0.75	0.4802	-0.0841	-0.0841	0.0375	0.0375	
0.80	0.5267	-0.0124	-0.0124	0.0954	0.0954	
0.85	0.5732	0.0626	0.0626	0.1546	0.1546	
0.90	0.6197	0.1409	0.1409	0.2151	0.2151	
0.95	0.6662	0.2222	0.2222	0.2766	0.2766	
0.98	0.6941	0.2723	0.2723	0.3140	0.3140	
0.99	0.7034	0.2892	0.2892	0.3266	0.3188	
1.00	0.7127	0.3088	0.3088	0.3391	0.3088	
Foil L. E. Circle	X = -0.1873	Y = -0.5144	R = 0.0100			
Foil T. E. Circle	X = 0.7027	Y = 0.3088				
Foil L. E. Tangency Pt. Suction	X = -0.2059	Y = -0.5379				
Foil L. E. Tangency Pt. Pressure	X = -0.1791	Y = -0.4856				
Foil T. E. Tangency Pt. Suction	X = 0.7115	Y = 0.3041				
Foil T. E. Tangency Pt. Pressure	X = 0.6947	Y = 0.3148				
Foil Nose Point	X = -0.2140	Y = -0.5007				
Foil Tail Point	X = 0.7081	Y = 0.3173				
Foil Area	= 0.1079					
Gaging Angle	LAMBDA = 0.4687 = 31.430	X = 0.3312	Y = 0.2881			
Center of Gravity Radial Reference	X = 0.2366 X = 0.3924	Y = -0.2680 Y = -0.3113				
Inlet Angle	= 62.900	DELTA BETA 1 = 22.500				
Exit Angle	= 32.570	DELTA BETA 2 = 8.300				
Uncovered Turning	= 10.575					
Min. Moment of Inertia	= 0.00023066					
Max. Moment of Inertia	= 0.01108872					
Principal Axis Angle	= 44.956					
Pitch/Axial Chord	= 0.9665					
Pitch/Actual Chord	= 0.7292					

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TABLE XXIV  
SECOND VANE CASCADE AIRFOIL, RECAMBERING DESIGN C

Radius of FF Section = 7.5000  
Pitch = 0.5890

Axial Chord = 0.8320  
No. of Foils = 80.00

Actual Chord = 0.8432

Percent X	Foil X	Foil Y Suction	Foil Y (Circle)	Foil Y Pressure	Foil Y (Circle)
0.0	0.0	-0.2161	-0.2105	-0.1601	-0.2105
0.01	0.0083	-0.2296	-0.2296	-0.1690	-0.1943
0.02	0.0166	-0.2424	-0.2424	-0.1778	-0.1908
0.03	0.0250	-0.2545	-0.2545	-0.1866	-0.1911
0.04	0.0333	-0.2659	-0.2659	-0.1954	-0.1955
0.05	0.0416	-0.2767	-0.2767	-0.2038	-0.2038
0.10	0.0832	-0.3223	-0.3223	-0.2329	-0.2329
0.15	0.1248	-0.3563	-0.3563	-0.2494	-0.2494
0.20	0.1664	-0.3808	-0.3808	-0.2584	-0.2584
0.25	0.2080	-0.3972	-0.3972	-0.2622	-0.2622
0.30	0.2496	-0.4065	-0.4065	-0.2618	-0.2618
0.35	0.2912	-0.4094	-0.4094	-0.2580	-0.2580
0.40	0.3328	-0.4066	-0.4066	-0.2512	-0.2512
0.45	0.3744	-0.3983	-0.3983	-0.2419	-0.2419
0.50	0.4160	-0.3849	-0.3849	-0.2301	-0.2301
0.55	0.4576	-0.3665	-0.3665	-0.2160	-0.2160
0.60	0.4992	-0.3435	-0.3435	-0.1998	-0.1998
0.65	0.5408	-0.3158	-0.3158	-0.1814	-0.1814
0.70	0.5824	-0.2834	-0.2834	-0.1609	-0.1609
0.75	0.6240	-0.2466	-0.2466	-0.1382	-0.1382
0.80	0.6656	-0.2055	-0.2055	-0.1133	-0.1133
0.85	0.7072	-0.1609	-0.1609	-0.0860	-0.0860
0.90	0.7488	-0.1137	-0.1137	-0.0562	-0.0562
0.95	0.7904	-0.0644	-0.0644	-0.0236	-0.0236
0.98	0.8154	-0.0340	-0.0340	-0.0025	-0.0025
0.99	0.8237	-0.0238	-0.0238	0.0047	-0.0001
1.00	0.8320	-0.0135	-0.0100	0.0119	-0.0100
Foil L.E. Circle	X = 0.0200	Y = -0.2105	R = 0.0200		
Foil T.E. Circle	X = 0.8220	Y = -0.0100	R = 0.0100		
Foil L.E. Tangency Pt. Suction	X = 0.0029	Y = -0.2209			
Foil L.E. Tangency Pt. Pressure	X = 0.0345	Y = -0.1968			
Foil T.E. Tangency Pt. Suction	X = 0.8298	Y = -0.0163			
Foil T.E. Tangency Pt. Pressure	X = 0.8154	Y = -0.0025			
Foil Nose Point	X = 0.0079	Y = -0.1946			
Foil Tail Point	X = 0.8289	Y = -0.0028			
Foil Area		= 0.0931			
Gaging	LAMBDA = 0.3860	X = 0.5732	Y = -0.2910		
Gaging Angle	= 40.940				
Center of Gravity	X = 0.3919	Y = -0.2691			
Radial Reference	X = 0.3919	Y = -0.2691			
Inlet Angle	= 37.330,	DELTA BETA 1 = 12.00			
Exit Angle	= 44.000	DELTA BETA 2 = 10.000			
Uncovered Turning	= 12.056				
Min. Moment of Inertia	= 0.00033518				
Max. Moment of Inertia	= 0.00401353				
Principal Axis Angle	= -16.359				
Pitch/Actual Chord	= 0.7080				
Pitch/Actual Chord	= 0.6988				

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TABLE XXV

## SECOND VANE CASCADE AIRFOIL, RECAMBERING DESIGN C

Radius of AA Section = 7.6500 Pitch = 0.6008		Axial Chord = 0.8357 No. of Foils = 80.00		Actual Chord - 0.8529	
Percent X	Foil X	Foil Y Suction	Foil Y (Circle)	Foil Y Pressure	Foil Y (Circle)
0.0	-0.0083	-0.2314	-0.2256	-0.1751	-0.2256
0.01	0.0001	-0.2448	-0.2448	-0.1838	-0.2092
0.02	0.0085	-0.2576	-0.2576	-0.1925	-0.2056
0.03	0.0168	-0.2697	-0.2697	-0.2011	-0.2058
0.04	0.0252	-0.2812	-0.2812	-0.2098	-0.2100
0.05	0.0335	-0.2921	-0.2921	-0.2181	-0.2181
0.10	0.0753	-0.3391	-0.3391	-0.2491	-0.2491
0.15	0.1171	-0.3750	-0.3750	-0.2692	-0.2692
0.20	0.1589	-0.4016	-0.4016	-0.2821	-0.2821
0.25	0.2007	-0.4202	-0.4202	-0.2894	-0.2894
0.30	0.2425	-0.4315	-0.4315	-0.2919	-0.2919
0.35	0.2842	-0.4363	-0.4363	-0.2901	-0.2901
0.40	0.3260	-0.4349	-0.4349	-0.2843	-0.2843
0.45	0.3678	-0.4278	-0.4278	-0.2748	-0.2748
0.50	0.4096	-0.4150	-0.4150	-0.2619	-0.2619
0.55	0.4514	-0.3968	-0.3968	-0.2458	-0.2458
0.60	0.4932	-0.3733	-0.3733	-0.2266	-0.2266
0.65	0.5350	-0.3443	-0.3443	-0.2047	-0.2047
0.70	0.5767	-0.3099	-0.3099	-0.1800	-0.1800
0.75	0.6185	-0.2698	-0.2698	-0.1526	-0.1526
0.80	0.6603	-0.2241	-0.2241	-0.1225	-0.1225
0.85	0.7021	-0.1735	-0.1735	-0.0896	-0.0896
0.90	0.7439	-0.1187	-0.1187	-0.0539	-0.0539
0.95	0.7857	-0.0609	-0.0609	-0.0152	-0.0152
0.98	0.8107	-0.0252	-0.0252	0.0096	0.0095
0.99	0.8191	-0.0132	-0.0132	0.0181	0.0120
1.00	0.8275	-0.0011	-0.0011	0.0265	0.0021
Foil L. E. Circle	X = 0.0121	Y = -0.2256	R = 0.0204		
Foil T. E. Circle	X = 0.8175	Y = 0.0021	R = 0.0100		
Foil L. E. Tangency Pt. Suction	X = -0.0052	Y = -0.2363			
Foil L. E. Tangency Pt. Pressure	X = 0.0268	Y = -0.2115			
Foil T. E. Tangency Pt. Suction	X = 0.8256	Y = -0.0037			
Foil T. E. Tangency Pt. Pressure	X = 0.8104	Y = 0.0092			
Foil Nose Point	X = -0.0004	Y = -0.2095			
Foil Tail Point	X = 0.8239	Y = 0.0097			
Foil Area	= 0.0948				
Gaging Angle	LAMBDA = 0.3659	X = 0.5704	Y = -0.3154		
Gaging Angle	= 37.511				
Center of Gravity, Radial Reference	X = 0.3932, X = 0.3919	Y = -0.2887 Y = -0.2691			
Inlet Angle	= 37.732	DELTA BETA 1 = 12.329			
Exit Angle	= 40.324	DELTA BETA 2 = 9.921			
Uncovered Turning	= 13.580				
Min. Moment of Inertia	= 0.00041249				
Max. Moment of Inertia	= 0.00430317				
Principal Axis Angle	= -17.910				
Pitch/Axial Chord	= 0.7189				
Pitch/Actual Chord	= 0.7044				

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TABLE XXVI

## SECOND VANE CASCADE AIRFOIL, RECAMBERING DESIGN C

Radius of BB Section = 8.4863 Pitch = 0.6665		Axial Chord = 0.8565 No. of Foils = 80.00		Actual Chord = 0.9216	
Percent X	Foil X	Foil Y Suction	Foil Y (Circle)	Foil Y Pressure	Foil Y (Circle)
0.0	-0.0543	-0.3164	-0.3093	-0.2604	-0.3093
0.01	-0.0458	-0.3286	-0.3286	-0.2677	-0.2917
0.02	-0.0372	-0.3406	-0.3406	-0.2751	-0.2875
0.03	-0.0286	-0.3521	-0.3521	-0.2824	-0.2871
0.04	-0.0201	-0.3633	-0.3633	-0.2898	-0.2901
0.05	-0.0115	-0.3741	-0.3741	-0.2971	-0.2971
0.10	0.0313	-0.4232	-0.4232	-0.3311	-0.3311
0.15	0.0741	-0.4641	-0.4641	-0.3605	-0.3605
0.20	0.1170	-0.4972	-0.4972	-0.3847	-0.3847
0.25	0.1598	-0.5230	-0.5230	-0.4031	-0.4031
0.30	0.2026	-0.5416	-0.5416	-0.4155	-0.4155
0.35	0.2454	-0.5530	-0.5530	-0.4212	-0.4212
0.40	0.2883	-0.5573	-0.5573	-0.4199	-0.4199
0.45	0.3311	-0.5544	-0.5544	-0.4115	-0.4115
0.50	0.3739	-0.5442	-0.5442	-0.3957	-0.3957
0.55	0.4167	-0.5264	-0.5264	-0.3727	-0.3727
0.60	0.4596	-0.5006	-0.5006	-0.3425	-0.3425
0.65	0.5024	-0.4661	-0.4661	-0.3056	-0.3056
0.70	0.5452	-0.4224	-0.4224	-0.2622	-0.2622
0.75	0.5880	-0.3684	-0.3684	-0.2128	-0.2128
0.80	0.6309	-0.3030	-0.3030	-0.1579	-0.1579
0.85	0.6737	-0.2259	-0.2259	-0.0981	-0.0981
0.90	0.7165	-0.1374	-0.1374	-0.0338	-0.0338
0.95	0.7593	-0.0390	-0.0390	0.0346	0.0346
0.98	0.7850	0.0242	0.0242	0.0773	0.0767
0.99	0.7936	0.0459	0.0459	0.0917	0.0796
1.00	0.8022	0.0678	0.0697	0.1061	0.0697
Foil L.E. Circle	X = -0.0318	Y = -0.3093	R = 0.0225		
Foil T.E. Circle	X = 0.7922	Y = 0.0697	R = 0.0100		
Foil L. E. Tangency Pt. Suction	X = -0.0503	Y = -0.3222			
Foil L. E. Tangency Pt. Pressure	X = -0.0172	Y = -0.2923			
Foil T. E. Tangency Pt. Suction	X = 0.8015	Y = 0.0661			
Foil T. E. Tangency Pt. Pressure	X = 0.7836	Y = 0.0748			
Foil Nose Point	X = -0.0469	Y = -0.2926			
Foil Tail Point	X = 0.7965	Y = 0.0787			
Foil Area	= 0.1053				
Gaging Angle	LAMBDA = 0.2911 = 25.900	X = 0.5594	Y = -0.4057		
Center of Gravity Radial Reference	X = 0.3893 X = 0.3919	Y = -0.3661 Y = -0.2691			
Inlet Angle	= 42.100	DELTA BETA 1 = 14.500			
Exit Angle	= 25.950	DELTA BETA 2 = 9.500			
Uncovered Turning	= 18.181				
Min. Moment of Inertia	= 0.00089655				
Max. Moment of Inertia	= 0.00611507				
Principal Axis Angle	= -27.285				
Pitch/Axial Chord	= 0.7782				
Pitch/Actual Chord	= 0.7232				

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TABLE XXVII

## SECOND VANE CASCADE AIRFOIL, RECAMBERING DESIGN C

Radius of CC Section = 9.4725 Pitch = 0.7440		Axial Chord = 0.8810 No. of Foils = 80.00		Actual Chord = 1.0247		
Percent X	Foil X	Foil Y Suction	Foil Y (Circle)	Foil Y Pressure	Foil Y (Circle)	
0.0	-0.1086	-0.4100	-0.4005	-0.3565	-0.4005	
0.01	-0.0998	-0.4199	-0.4199	-0.3618	-0.3814	
0.02	-0.0910	-0.4296	-0.4296	-0.3670	-0.3766	
0.03	-0.0822	-0.4391	-0.4391	-0.3722	-0.3755	
0.04	-0.0734	-0.4484	-0.4484	-0.3775	-0.3777	
0.05	-0.0646	-0.4575	-0.4575	-0.3827	-0.3827	
0.10	-0.0205	-0.5000	-0.5000	-0.4077	-0.4077	
0.15	0.0235	-0.5371	-0.5371	-0.4305	-0.4305	
0.20	0.0676	-0.5687	-0.5687	-0.4506	-0.4506	
0.25	0.1116	-0.5943	-0.5943	-0.4674	-0.4674	
0.30	0.1557	-0.6136	-0.6136	-0.4800	-0.4800	
0.35	0.1997	-0.6263	-0.6263	-0.4876	-0.4876	
0.40	0.2438	-0.6320	-0.6320	-0.4889	-0.4889	
0.45	0.2878	-0.6301	-0.6301	-0.4828	-0.4828	
0.50	0.3319	-0.6200	-0.6200	-0.4678	-0.4678	
0.55	0.3759	-0.6011	-0.6011	-0.4428	-0.4428	
0.60	0.4200	-0.5726	-0.5726	-0.4071	-0.4071	
0.65	0.4640	-0.5335	-0.5335	-0.3605	-0.3605	
0.70	0.5081	-0.4827	-0.4827	-0.3034	-0.3034	
0.75	0.5521	-0.4187	-0.4187	-0.2368	-0.2368	
0.80	0.5962	-0.3400	-0.3400	-0.1620	-0.1620	
0.85	0.6402	-0.2448	-0.2448	-0.0801	-0.0801	
0.90	0.6843	-0.1320	-0.1320	0.0074	0.0074	
0.95	0.7283	-0.0010	-0.0010	0.0996	0.0996	
0.98	0.7547	0.0863	0.0863	0.1568	0.1559	
0.99	0.7635	0.1168	0.1168	0.1760	0.1593	
1.00	0.7724	0.1480	0.1480	0.1953	0.1494	
Foil L.E. Circle	X = -0.0836	Y = -0.4005	R = 0.0250			
Foil T.E. Circle	X = 0.7624	Y = 0.1494	R = 0.0100			
Foil L. E. Tangency Pt. Suction	X = -0.1023	Y = -0.4171				
Foil L. E. Tangency Pt. Pressure	X = -0.0709	Y = -0.3790				
Foil T. E. Tangency Pt. Suction	X = 0.7720	Y = 0.1467				
Foil T. E. Tangency Pt. Pressure	X = 0.7533	Y = 0.1536				
Foil Nose Point	X = -0.1029	Y = -0.3846				
Foil Tail Point	X = 0.7658	Y = 0.1588				
Foil Area	= 0.1192					
Gaging	LAMBDA = 0.2620	X = 0.5362	Y = -0.4434			
Gaging Angle	= 20.620					
Center of Gravity	X = 0.3649	Y = -0.4037				
Radial Reference	X = 0.3919	Y = -0.2691				
Inlet Angle	= 50.500	DELTA BETA 1 = 17.500				
Exit Angle	= 20.100	DELTA BETA 2 = 9.000				
Uncovered Turning	= 18.178					
Min Moment of Inertia	= 0.00126226					
Max. Moment of Inertia	= .00876554					
Principal Axis Angle	= -35.891					
Pitch/Axial Chord	= 0.8445					
Pitch/Actual Chord	= 0.7261					

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TABLE XXVIII  
SECOND VANE CASCADE AIRFOIL, RECAMBERING DESIGN C

Radius of DD Section = 10.4588 Pitch = 0.8214		Axial Chord = 0.9055 No. of Foils = 80.00		Actual Chord = 1.1187		
Percent X	Foil X	Foil Y Suction	Foil Y (Circle)	Foil Y Pressure	Foil Y (Circle)	
0.0	-0.1630	-0.4600	-0.4479	-0.4067	-0.4479	
0.01	-0.1539	-0.4683	-0.4683	-0.4104	-0.4275	
0.02	-0.1449	-0.4765	-0.4765	-0.4142	-0.4221	
0.03	-0.1358	-0.4845	-0.4845	-0.4180	-0.4204	
0.04	-0.1268	-0.4924	-0.4924	-0.4217	-0.4218	
0.05	-0.1177	-0.5000	-0.5000	-0.4255	-0.4255	
0.10	-0.0724	-0.5350	-0.5350	-0.4431	-0.4431	
0.15	-0.0272	-0.5646	-0.5646	-0.4585	-0.4585	
0.20	0.0181	-0.5886	-0.5886	-0.4711	-0.4711	
0.25	0.0634	-0.6067	-0.6067	-0.4804	-0.4804	
0.30	0.1037	-0.6187	-0.6187	-0.4854	-0.4854	
0.35	0.1539	-0.6241	-0.6241	-0.4851	-0.4851	
0.40	0.1992	-0.6225	-0.6225	-0.4785	-0.4785	
0.45	0.2445	-0.5134	-0.6134	-0.4642	-0.4642	
0.50	0.2898	-0.5963	-0.5963	-0.4409	-0.4409	
0.55	0.3350	-0.5704	-0.5704	-0.4076	-0.4076	
0.60	0.3803	-0.5349	-0.5349	-0.3639	-0.3639	
0.65	0.4256	-0.4889	-0.4889	-0.3097	-0.3097	
0.70	0.4709	-0.4311	-0.4311	-0.2459	-0.2459	
0.75	0.5161	-0.3599	-0.3599	-0.1735	-0.1735	
0.80	0.5614	-0.2738	-0.2738	-0.0938	-0.0938	
0.85	0.6067	-0.1717	-0.1717	-0.0081	-0.0081	
0.90	0.6520	-0.0535	-0.0535	0.0826	0.0826	
0.95	0.6972	0.0801	0.0801	0.1771	0.1771	
0.98	0.7244	0.1671	0.1671	0.2354	0.2350	
0.99	0.7335	0.1971	0.1971	0.2550	0.2391	
1.00	0.7425	0.2277	0.2277	0.2746	0.2291	
Foil L.E. Circle	X = -0.1355	Y = -0.4479	R = 0.0275			
Foil T.E. Circle	X = 0.7325	Y = 0.2291	R = 0.0100			
Foil L.E. Tangency Pt. Suction	X = -0.1541	Y = -0.4681				
Foil L.E. Tangency Pt. Pressure	X = -0.1249	Y = -0.4225				
Foil T.E. Tangency Pt. Suction	X = 0.7421	Y = 0.2263				
Foil T.E. Tangency Pt. Pressure	X = 0.7234	Y = -0.2333				
Foil Nose Point	X = -0.1586	Y = -0.4331				
Foil Tail Point	X = 0.7360	Y = 0.2385				
Foil Area	= 0.1236					
Gaging	LAMBDA = 0.2969	X = 0.4789	Y = -0.4194			
Gaging Angle	= 21.190					
Center of Gravity	X = 0.3240	Y = -0.3762				
Radial Reference	X = 0.3919	Y = -0.2691				
Inlet Angle	= 57.400	DELTA BETA 1 = 20.000				
Exit Angle	= 20.600	DELTA BETA 2 = 8.400				
Uncovered Turning	= 17.802					
Min. Moment of Inertia	= 0.00110931					
Max. Moment of Inertia	= 0.01076076					
Principal Axis Angle	= 40.470					
Pitch/Axial Chord	= 0.9072					
Pitch/Actual Chord	= 0.7343					

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TABLE XXIX  
SECOND VANE CASCADE AIRFOIL, RECAMBERING DESIGN C

Radius of EE Section = 10.8600 Pitch = 0.8529		Axial Chord = 0.9155 No. of Foils - 80.00	Actual Chord = 1.1438		
Percent X	Foil X	Foil Y Suction	Foil Y (Circle)	Foil Y Pressure	Foil Y (Circle)
0.0	-0.1851	-0.4552	-0.4422	-0.4016	-0.4422
0.01	-0.1759	-0.4631	-0.4631	-0.4049	-0.4212
0.02	-0.1668	-0.4709	-0.4709	-0.4082	-0.4155
0.03	-0.1576	-0.4783	-0.4783	-0.4114	-0.4137
0.04	-0.1485	-0.4854	-0.4854	-0.4147	-0.4148
0.05	-0.1393	-0.4923	-0.4923	-0.4180	-0.4180
0.10	-0.0935	-0.5219	-0.5219	-0.4315	-0.4315
0.15	-0.0478	-0.5442	-0.5442	-0.4406	-0.4406
0.20	-0.0020	-0.5598	-0.5598	-0.4452	-0.4452
0.25	0.0438	-0.5686	-0.5686	-0.4453	-0.4453
0.30	0.0896	-0.5708	-0.5708	-0.4405	-0.4405
0.35	0.1353	-0.5664	-0.5664	-0.4304	-0.4304
0.40	0.1811	-0.5554	-0.5554	-0.4144	-0.4144
0.45	0.2269	-0.5375	-0.5375	-0.3917	-0.3917
0.50	0.2727	-0.5123	-0.5123	-0.3619	-0.3619
0.55	0.3184	-0.4798	-0.4798	-0.3242	-0.3242
0.60	0.3642	-0.4392	-0.4392	-0.2784	-0.2784
0.65	0.4100	-0.3900	-0.3900	-0.2248	-0.2248
0.70	0.4558	-0.3312	-0.3312	-0.1640	-0.1640
0.75	0.5015	-0.2619	-0.2619	-0.0968	-0.0968
0.80	0.5473	-0.1809	-0.1809	-0.0244	-0.0244
0.85	0.5931	-0.0876	-0.0876	0.0525	0.0525
0.90	0.6388	0.0176	-0.0176	0.1329	0.1329
0.95	0.6846	0.1338	0.1338	0.2163	0.2163
0.98	0.7121	0.2083	0.2083	0.2674	0.2671
0.99	0.7212	0.2338	0.2338	0.2846	0.2715
1.00	0.7304	0.2597	0.2616	0.3018	0.2616
Foil L.E. Circle	X = -0.1566	Y = -0.4422	R = 0.0285		
Foil T.E. Circle	X = 0.7204	Y = 0.2616	R = 0.0100		
Foil L.E. Tangency Pt. Suction	X = -0.1752	Y = -0.4638			
Foil L.E. Tangency Pt. Pressure	X = -0.1469	Y = -0.4153			
Foil T.E. Tangency Pt. Suction	X = 0.7297	Y = 0.2580			
Foil T.E. Tangency Pt. Pressure	X = 0.7117	Y = 0.2664			
Foil Nose Point	X = -0.1812	Y = -0.4278			
Foil Tail Point	X = 0.7246	Y = 0.2706			
Foil Area	= 0.1167				
Gaging Angle	LAMBDA = 0.3579 = 24.910	X = 0.4308	Y = -0.3645		
Center of Gravity Radial Reference	X = 0.2956 X = 0.3919	Y = -0.3218 Y = -0.2691			
Inlet Angle	= 59.762	DELTA BETA 1 = 23.993			
Exit Angle	= 25.141,	DELTA BETA 2 = 3.203			
Uncovered Turning	= 17.109				
Min. Moment of Inertia	= 0.00073264				
Max. Moment of Inertia	= 0.01057891				
Principal Axis Angle	= -40.621				
Pitch/Axial Chord	= 0.9317				
Pitch/Actual Chord	= 0.7457				

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TABLE XXX

## SECOND VANE CASCADE AIRFOIL, RECAMBERING DESIGN C

Radius of HH Section = 11.4100  
 Pitch = 0.8961

Axial Chord = 0.9291  
 No. of Foils = 80.00

Actual Chord = 1.1616

Percent X	Foil X	Foil Y Suction	Foil Y (Circle)	Foil Y Pressure	Foil Y (Circle)
0.0	-0.2154	-0.4210	-0.4066	-0.3668	-0.4066
0.01	-0.2061	-0.4282	-0.4282	-0.3695	-0.3849
0.02	-0.1968	-0.4355	-0.4355	-0.3722	-0.3789
0.03	-0.1875	-0.4421	-0.4421	-0.3748	-0.3767
0.04	-0.1782	-0.4480	-0.4480	-0.3775	-0.3776
0.05	-0.1689	-0.4535	-0.4535	-0.3800	-0.3800
0.10	-0.1225	-0.4725	-0.4725	-0.3859	-0.3859
0.15	-0.0760	-0.4796	-0.4796	-0.3817	-0.3817
0.20	-0.0295	-0.4767	-0.4767	-0.3686	-0.3686
0.25	0.0169	-0.4651	-0.4651	-0.3481	-0.3481
0.30	0.0634	-0.4459	-0.4459	-0.3215	-0.3215
0.35	0.1098	-0.4201	-0.4201	-0.2900	-0.2900
0.40	0.1563	-0.3884	-0.3884	-0.2545	-0.2545
0.45	0.2027	-0.3514	-0.3514	-0.2154	-0.2154
0.50	0.2492	-0.3095	-0.3095	-0.1737	-0.1737
0.55	0.2956	-0.2634	-0.2634	-0.1295	-0.1295
0.60	0.3421	-0.2133	-0.2133	-0.0833	-0.0833
0.65	0.3886	-0.1595	-0.1595	-0.0351	-0.0351
0.70	0.4350	-0.1021	-0.1021	0.0145	0.0145
0.75	0.4815	-0.0415	-0.0415	0.0655	0.0655
0.80	0.5279	0.0221	0.0221	0.1175	0.1175
0.85	0.5744	0.0887	0.0887	0.1705	0.1705
0.90	0.6208	0.1579	0.1579	0.2241	0.2241
0.95	0.6673	0.2295	0.2295	0.2785	0.2785
0.98	0.6952	0.2734	0.2734	0.3114	0.3114
0.99	0.7045	0.2882	0.2882	0.3224	0.3160
1.00	0.7138	0.3031	0.3060	0.3334	0.3060
Foil L. E. Circle	X = -0.1855	Y = -0.4066	R = 0.0299		
Foil T. E. Circle	X = 0.7036	Y = 0.3060	R = 0.0100		
Foil L. E. Tangency Pt. Suction	X = -0.2041	Y = -0.4300			
Foil L. E. Tangency Pt. Pressure	X = -0.1772	Y = -0.3778			
Foil T. E. Tangency Pt. Suction	X = 0.7122	Y = 0.3006			
Foil T. E. Tangency Pt. Pressure	X = 0.6962	Y = 0.3125			
Foil Nose Point	X = -0.2120	Y = -0.3929			
Foil Tail Point	X = 0.7097	Y = 0.3140			
Foil Area	= 0.0965				
Gaging Angle	LAMBDA = 0.5111 = 34.771	X = 0.3212	Y = -0.2364		
Center of Gravity Radial Reference	X = 0.2328 X = 0.3919	Y = -0.2084 Y = -0.2694			
Inlet Angle	= 62.722	DELTA BETA 1 = 22.408			
Exit Angle	= 36.630	DELTA BETA 2 = 8.009			
Uncovered Turning	= 10.092				
Min. Moment of Inertia	= 0.0002093				
Max. Moment of Inertia	= 0.00895742				
Principal Axis Angle	= -40.993				
Pitch/Axial Chord	= 0.9645				
Pitch/Actual Chord	= 0.7715				

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TABLE XXXI  
SECOND VANE CASCADE AIRFOIL, RECAMBERING DESIGN C

Radius of GG Section = 11.4450 Pitch = 0.8989	Axial Chord = 0.9300 No. of Foils = 80.00	Actual Chord = 1.1621			
Percent X	Foil X	Foil Y Suction	Foil Y (Circle)	Foil Y Pressure	Foil Y (Circle)
0.0	-0.2173	-0.4177	-0.4032	-0.3635	-0.4032
0.01	-0.2080	-0.4250	-0.4249	-0.3661	-0.3814
0.02	-0.1987	-0.4322	-0.4322	-0.3687	-0.3754
0.03	-0.1894	-0.4387	-0.4387	-0.3714	-0.3732
0.04	-0.1801	-0.4446	-0.4446	-0.3740	-0.3740
0.05	-0.1708	-0.4499	-0.4499	-0.3765	-0.3765
0.10	-0.1243	-0.4681	-0.4681	-0.3818	-0.3818
0.15	-0.0778	-0.4740	-0.4740	-0.3766	-0.3766
0.20	-0.0313	-0.4697	-0.4697	-0.3621	-0.3621
0.25	0.0152	-0.4565	-0.4565	-0.3399	-0.3399
0.30	0.0617	-0.4356	-0.4356	-0.3117	-0.3117
0.35	0.1082	-0.4081	-0.4081	-0.2785	-0.2785
0.40	0.1547	-0.3748	-0.3748	-0.2414	-0.2414
0.45	0.2012	-0.3363	-0.3363	-0.2012	-0.2012
0.50	0.2477	-0.2932	-0.2932	-0.1585	-0.1585
0.55	0.2942	-0.2460	-0.2460	-0.1139	-0.1139
0.60	0.3407	-0.1951	-0.1951	-0.0677	-0.0677
0.65	0.3872	-0.1410	-0.1410	-0.0201	-0.0201
0.70	0.4337	-0.0838	-0.0838	0.0285	0.0285
0.75	0.4802	-0.0241	-0.0241	0.0781	0.0781
0.80	0.5267	0.0381	0.0381	0.1284	0.1284
0.85	0.5732	0.1024	0.1024	0.1793	0.1793
0.90	0.6197	0.1685	0.1685	0.2309	0.2309
0.95	0.6662	0.2364	0.2364	0.2828	0.2828
0.98	0.6941	0.2779	0.2779	0.3142	0.3142
0.99	0.7034	0.2918	0.2918	0.3247	0.3188
1.00	0.7127	0.3058	0.3088	0.3352	0.3088
Foil L.E. Circle	X = -0.1873	Y = -0.4032	R = 0.0300		
Foil T.E. Circle	X = 0.7027	Y = 0.3088	R = 0.0100		
Foil L.E. Tangency Pt. Suction	X = -0.2059	Y = -0.4267			
Foil L.E. Tangency Pt. Pressure	X = -0.1791	Y = -0.3743			
Foil T.E. Tangency Pt. Suction	X = 0.7110	Y = 0.3033			
Foil T.E. Tangency Pt. Pressure	X = 0.6952	Y = 0.3155			
Foil Nose Point	X = -0.2140	Y = -0.3895			
Foil Tail Point	X = 0.7088	Y = 0.3168			
Foil Area	0.0947				
Gaging Gaging Angle	LAMBDA = 0.5234 35.610	X = 0.3107	Y = -0.2283		
Center of Gravity Radial Reference	X = 0.2272 X = 0.3919	Y = -0.2003 Y = -0.2691			
Inlet Angle Exit Angle Uncovered Turning	= 62.900 = 30.570 = 9.150	DELTA BETA 1 = 22.500 DELTA BETA 2 = 8.000			
Min. Moment of Inertia Max. Moment of Inertia	= 0.00018734 = 0.00878485				
Principal Axis	= -41.099				
Pitch/Axial Chord Pitch/Actual Chord	= 3.9665 = 0.7735				

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## **REFERENCES**

1. H. Welna, D. E. Dahlberg and W. H. Heiser; (Unclassified title) Investigation of a Highly Loaded Two-Stage Fan-Drive Turbine; Technical Report AFAPL-TR-69-92 Volume 1; Pratt & Whitney Aircraft Division of United Aircraft Corporation; June 1968; Confidential.
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(U) A comprehensive, four-phase, three-year program is in progress to investigate methods of improving the performance of fan-drive turbines. The goals of this program are to develop turbine design procedures and aerodynamic techniques for efficient, high work, low pressure turbines.

(U) The first phase effort of defining the preliminary turbine design has been completed and the results were reported (Reference 1). The second phase consisted of an experimental evaluation which included establishment of both two-dimensional loss levels and three-dimensional flow behavior for the baseline airfoils, and for airfoils utilizing various boundary layer control methods. The design of the baseline cascade pack was reported in Reference 2, and the annular cascade performance of the baseline airfoils was reported in Reference 3. The annular cascade results of the baseline airfoil boundary layer control methods were reported in Reference 4.

(U) The results of the plane-cascade, two-dimensional, baseline investigation, the decreased-solidity annular-cascade airfoil test results, and the Phase III boundary layer control optimization and off-design evaluation are presented in this report.

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